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SLV-IB-3

HEADQUARTERS
AIR FORCE SPECIAL WEAPONS CENTER
AIR FORCE SYSTEMS COMMAND
KIRTLAND AIR FORCE BASE, NEW MEXICO

USERS GUIDE TO PAYLOAD PLANNING FOR THE SLV-IB SPACE PROBE

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SPACE VEHICLES BRANCH
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January 1963

TO THE HOLDER OF THIS GUIDE:

If you will return a note to Hq AFSWC (SWTTS), Kirtland AFB, NMex, with your name and correct mailing address, we will forward any changes to the manual directly to you. If we can be of further assistance, do not hesitate to call. Our phone number is given on page 78.

This report reviews the considerations presented to an experimenter in considering or using the SLV-1B as a launching vehicle. Included are discussion of payload weights, apogees, environments, volumes, launch facilities, documentation, suggested schedules, and applicable agencies in the SLV-1B program. The SLV-1B is a relatively low cost high performance space probe, used principally for conducting relatively lightweight experiments in the region beyond 10,000 n.m. or for conducting preliminary test flights for satellite and deep space missions. The SLV-1B program utilizes an all-USAF planning, procuring, documenting, technical consulting, and launching organization.

This report is prepared in recognition of the fact that the success of a deep space probe program is directly related to the degree to which all participating agencies understand and cooperate in all phases of the operation.

PUBLICATION REVIEW

This report has been reviewed and is approved.

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SLV-1B USERS' GUIDE

INDEX

I. VEHICLE DESCRIPTION.

A. Vehicle Configuration. The Standard Launch Vehicle-1B (SLV-1B) is a four-stage solid propellant unguided spin stabilized space probe. Physical makeup (and terminology associated with some of the inter-staging) of the vehicle is shown in Figure 1. The first stage motor is a Thiokol XM-33E7-3 (Castor). The second stage is an ABL X-259 (Antares). The third stage is an Aerojet General 30KS8000 (AJ-10). The fourth stage motor is a 17-inch spherical NOTS 100A (ARC improved). Stages 2, 3, and 4 are connected to the preceding stages by blowout diaphragms. First stage ignition is by blockhouse sequencer. Second stage ignition is actuated by a first-stage pressure switch in series with a timer. The third stage is fired by a second-stage headcap pressure switch and the fourth stage is similarly ignited by a third-stage headcap pressure switch. The first stage normally burns for about thirty-seven seconds followed by a coast period to an altitude of about 180,000 feet. The heatshield is then ejected and the second stage ignited. The second stage then burns for about thirty-four seconds, the third stage for twenty-seven seconds, and the fourth stage for about thirty-six seconds (See curves of longitudinal acceleration versus time, Section II, E.2.).

Normally, a command-destruct system is carried which can be activated through second-stage burning, destroying the first and second stages and preventing ignition of the third and fourth stages.

A Motorola SST-131 radar transponder is located in upper H section, allowing radar track through third-stage burning.

The fourth stage carries a radar reflector system, which, when tracked by the AN/FPQ-6 radar, allows tracking of the fourth stage to burnout.

Additional tracking, for the remainder of the flight, is normally by means of TLM-18 60-foot parabolic antenna telemetry track.

Figure 2 shows a SLV-1B in launch position. Initially, the launcher is positioned in azimuth and elevation, with corrections for wind weighting to insure that the proper flight corridor will be followed. At ignition, the missile moves forward on guide rails. The spin rocket assembly is fired by lanyard pull, once the missile has cleared the launcher rail. These rockets impact an initial spin to the vehicle to assist in achieving low-speed aerodynamic stability. First-stage fins maintain this spin rate as velocity increases.

SLV-1E ROCKET VEHICLE

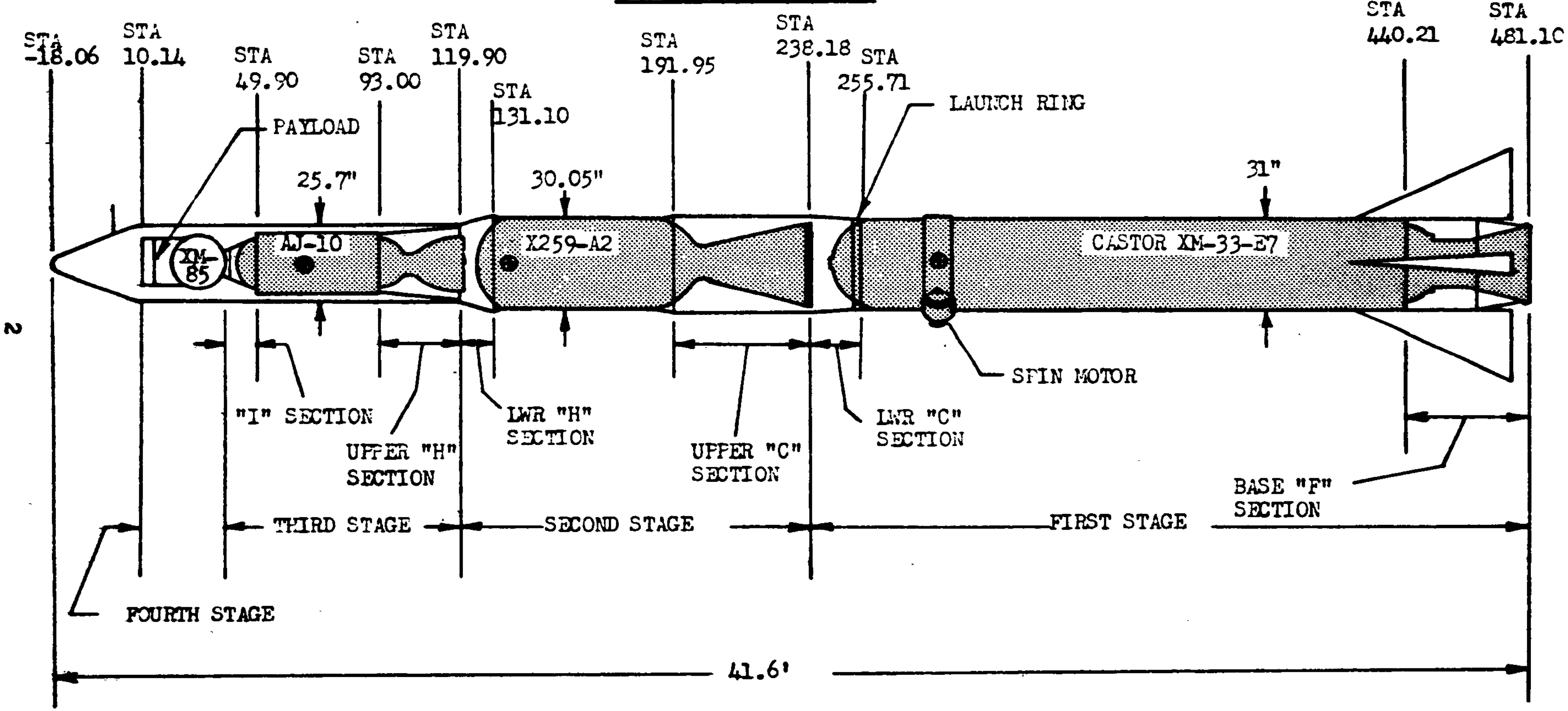
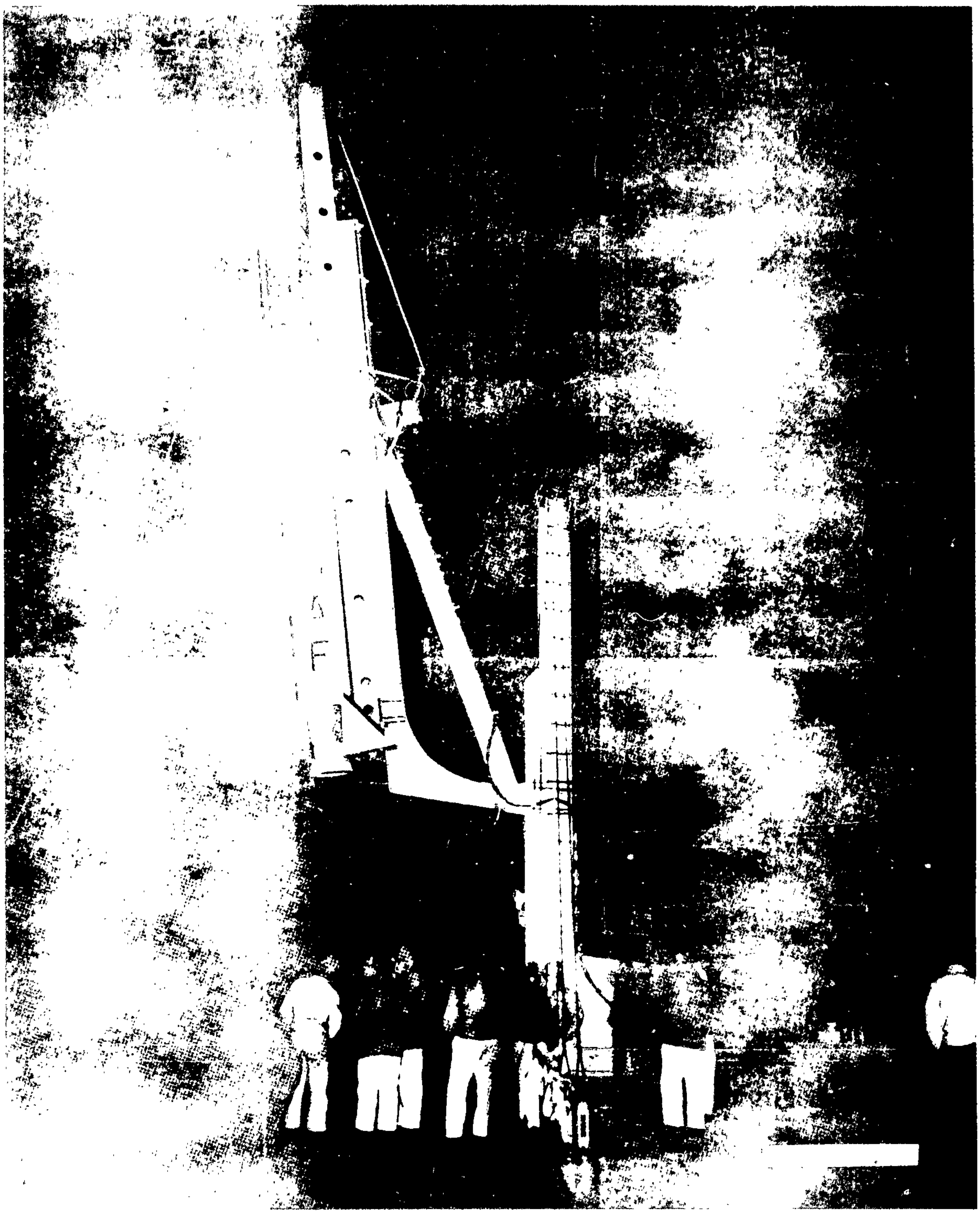


Figure 1.



B. <u>Apogee and Time of Flight Versus Payload Weight</u>.

Performance of the SLV-1B changes with several independent variables. Chief among these variables are payload weight, launch elevation angle, launch site and coasting time between stage ignitions. Most controllable of these is payload weight. Figure 3 presents a nominal performance curve of apogee versus payload weight and shows typical time of flight for various apogees. For particular payload weights, specific trajectories are computed by AFSWC (SWTTS). Normal launches from AMR will be on an azimuth of 105°.

The SLV-1B cannot place a payload in orbit because it does not contain a guidance system necessary for the application of thrust along the vector required for injection. However, as shown in the performance curves, escape velocities can be achieved with payloads of sufficiently light weight.

C. <u>Trajectory Computation and Accuracy</u>.

Following fourth-stage burnout, the payload and spent fourth-stage motor casing enter a free ballistic trajectory. This trajectory can be predicted nominally to within 10% of apogee altitude by computer runs using correct payload and motor weights. These trajectories are, for various reasons, not included in this manual. To establish in-flight trajectory information, radar track through fourth-stage burnout is required. After-the-fact-trajectories and real-time acquisition data are normally provided by TLM-18 azimuth-elevation telemetry track. Azimuths and elevations from several stations are triangulated to establish points on the trajectory. A ballistic curve, fitted to these points, normally gives trajectory determination to within 2% accuracy.

D. <u>Alternate Launch Vehicles</u>.

If, after due consideration, it is found that the SLV-1B space probe will not accomplish the desired mission objectives, consideration should be directed toward other configurations. Other vehicles presently in the AF inventory include the guided SLV-1 series, the unguided Nike-Cajun, Nike-Yardbird, Aerobee 150, Javelin, the Astrobee 200, and "piggy-back" scientific passenger pods. Three and two-stage versions of the SLV-1B also exist, with a capability of boosting 250 pounds to 1200 n.m., and 250 pounds to 300 n.m., respectively.

All agencies in the SLV-1B program will be happy to assist any experimenter in determining the most suitable vehicle for a particular mission.

SLV - 1B (As of 23 Dec 1962)

Figure 3.

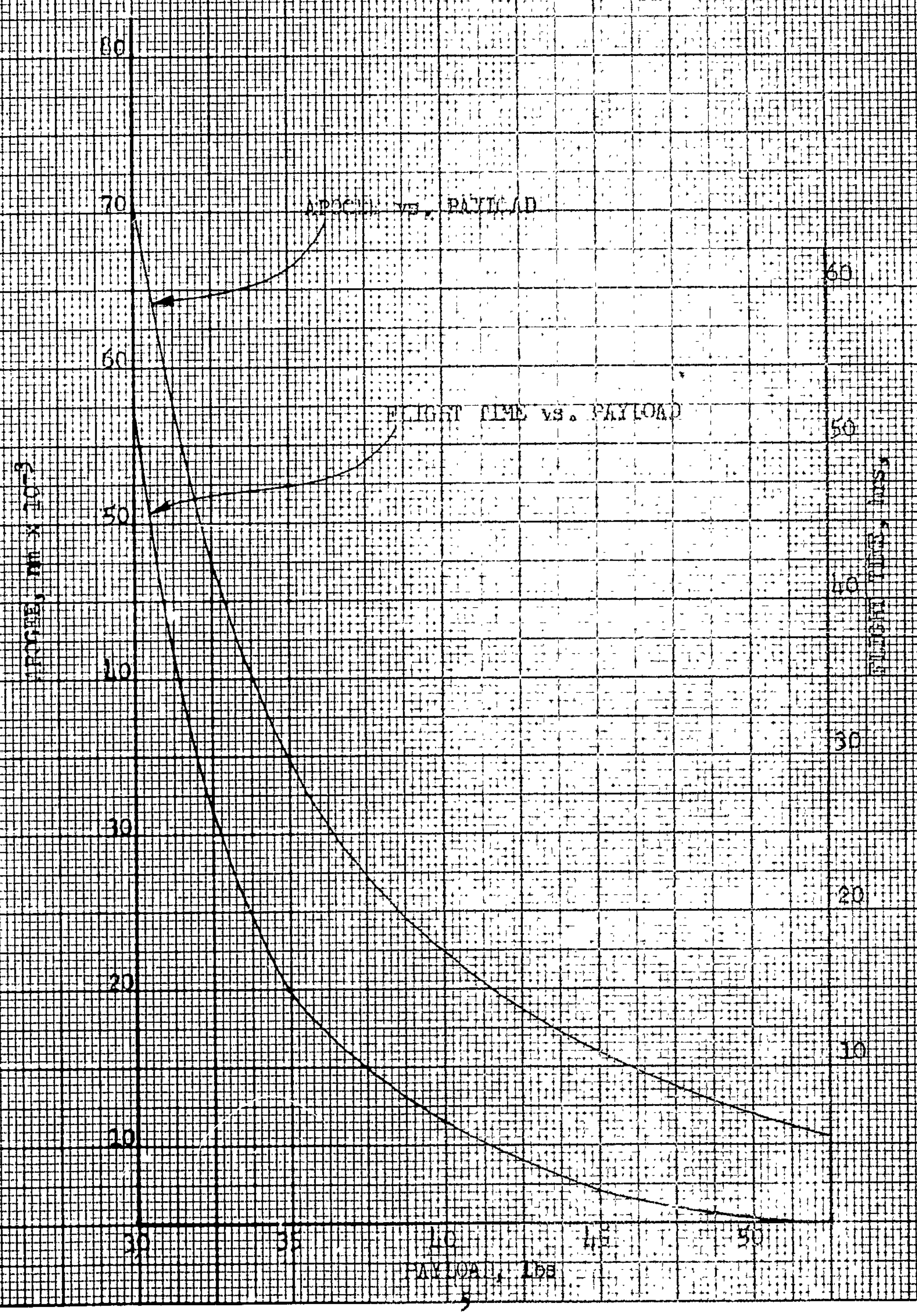
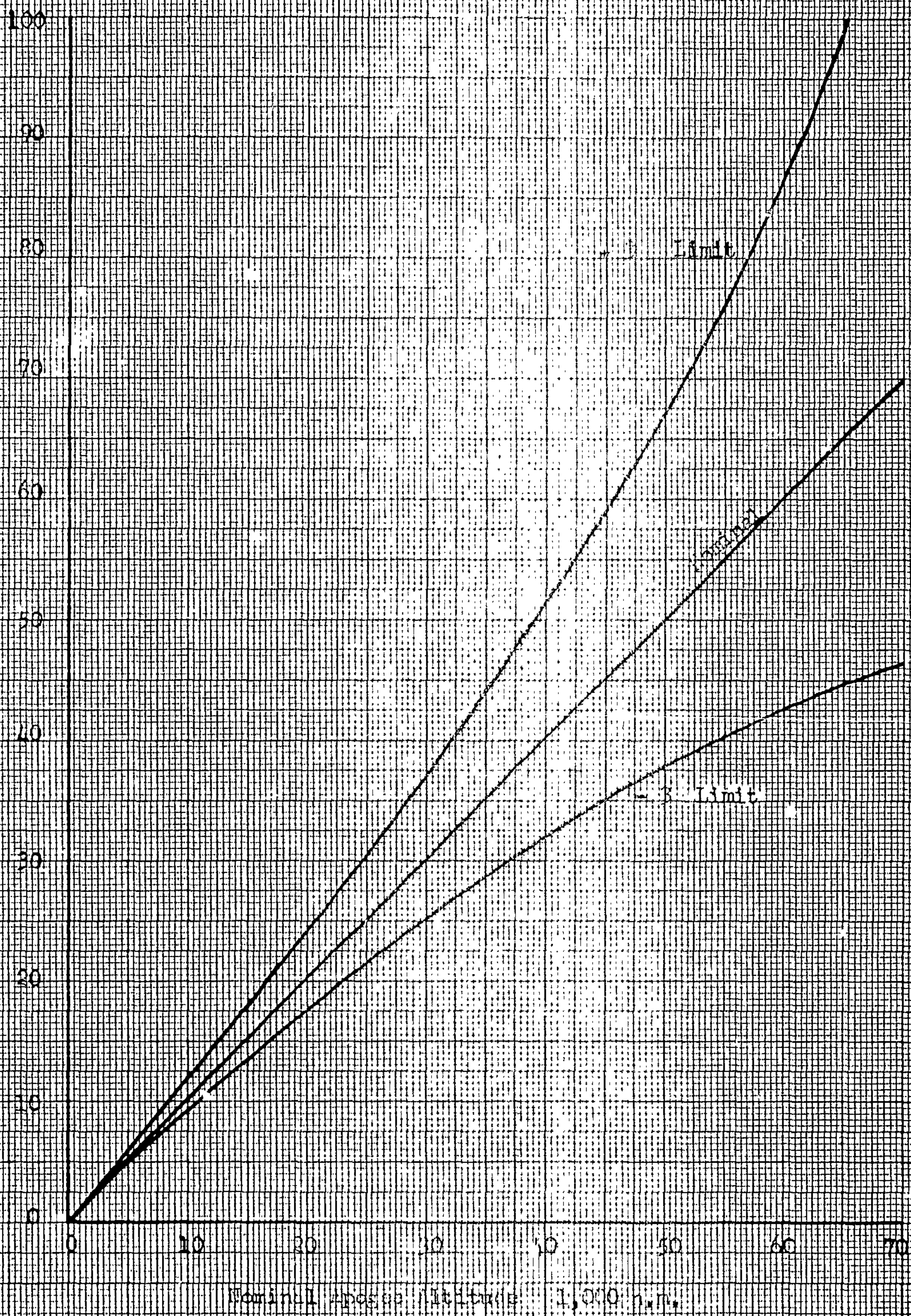


Figure 4.

APOGEE ALTITUDE DISPERSION



II. CONSIDERATIONS IN PLANNING A PAYLOAD FOR THE SLV-1B VEHICLE.

A. Arriving at a Payload Weight. Perhaps the most important single question to be answered, in examining the SLV-1B as a booster, is: Are the payload weight and desired altitude within the performance capabilities of the booster?

1. To approach this question, the experimenter must first define his experiment closely, particularly in terms of minimum altitude required for successful accomplishment of the experiment and time required above a certain altitude. If a tabulated trajectory is required, this may be requested from AFSWC (SWTTS) by specifying desired apogee, or expected payload weight.

2. Once the experiment has been defined, referral to the chart of apogee versus payload weight will establish the maximum allowable payload weight necessary to achieve the required altitude. Note that this is maximum allowable payload weight. To provide a confidence factor, it is strongly recommended that 95% maximum allowable payload weight be chosen as the design maximum.

3. Can the experiment be performed within the design maximum weight constraints? Estimate weight of experiment, spacecraft structure, telemetry system and required battery weight. (A table of battery weights is given in Appendix A; of typical component weights in Appendix C.) Required telemetry system power consumption (and, hence battery weight) may be reduced in several ways:

a. Reduce the data requirements (i.e., transmit fewer bit/sec).

b. Use a more efficient system (in terms of power consumed per bit transmitted).

If the experiment, as defined in 1, above, can be flown successfully in the SLV-1B vehicle, a workable compromise between battery weight and information rate can be found. Within the confines thus established, payload design can begin.

B. Selection of a Telemetering System.

Choice of the type of telemetering system to be used is extremely important. Due to present National Missile Range tracking station capability, the RF frequency must be in the 215-260 MC band. Frequency allocations will be requested by AFSWC (SWTTS). As has been demonstrated in paragraph A, the transmitted bit rate and power efficiency of the system play a determining part in assessing whether an experiment can be flown in the SLV-1B.

Recommended telemetering system types are the following: modified FM/FM (reduced bandwidth), PAM/FM, FM, PCM/PM. Each has attributes which make it worth considering. The final choice of a system depends upon the constraints of the particular experiment. Consulting service in this area, if desired, is available from AFSWC (SWTTS).

The required transmitter power may be calculated from the following equation:

$$P_t = \frac{4\pi R^2 (NF) KT (S/N) B}{G_t \; A_R}$$

where

R = range in feet from transmitter to receiver.

(NF) = system noise figure (4.5 db = 2.82 is a good estimate).

KT = $1.37 \times 10^{-23}(290) = 4.05 \times 10^{-21}$.

S/N = required predetection signal-to-noise ratio (varies from + 20 db to - 6 db, depending upon the system).

B = receiver bandwidth in cycles/second (should coincide, if possible, with National Missile Range standard receiver bandwidths). Note: This factor is determined directly by the transmitted data rate.

G_t = gain of transmitting antenna (This requires some knowledge of what the spacecraft antenna patterns are like; nulls may generally be held to - 3 db or less.)

A_R = effective area of receiving antenna in square feet (850 ft^2 for a 60-foot diameter TLM-18 type dish at 250 MC).

A communications efficiency comparison may be made by substituting $\beta H = (S/N)B$ in the preceding equation.

β is a measure of communications efficiency.*

$$\beta = \frac{P_R}{\emptyset H}$$

P_R = required received signal power (watts)

$\emptyset$ = noise spectral density (watts/cycle)

H = information rate (bits/sec)

A system of high communications efficiency, capable of transmitting 64 bits of information per second from 50,000 miles at a 10^{-6} error rate with a one-quarter watt transmitter, has been developed for AFSWC. This system was flown on 4 Dec 1961. Special receivers for this system will be operational at the Atlantic Missile Range in 1963. (See Appendix B for further information on this system.)

Choice of a telemetering system must be made at the very beginning of the design phase, since sensor and electronic outputs must be made compatible with telemetering system inputs. In addition, at least six months procurement time should be allowed from RFQ to delivery of a system. If the system is non-Range standard, a minimum of 18 months in advance of launch must be allowed for the Range to procure suitable support equipment. At least 9 months before launch, all particulars of the system must be known if more than one National Missile Range is to provide tracking and data gathering support. The transmitter should comply with interference standards IRIG 106-60 and MIL-I-26600. An RF spectrum analysis on the transmitter (or the transmitter itself) <u>must</u> be submitted to the Range at the time of flight documentation if a spectrum analysis on the particular transmitter is not on file at the Range.

C. <u>Prime Power Considerations</u>.

For most SLV-1B payloads, the lightest (in terms of watt hours/pound) means of supplying energy is silver-zinc batteries. Tables of one manufacturer's silver-zinc cells are given in Appendix A. Tests and repeated use of these primary cells have shown them to be entirely reliable and well-suited to space probe applications.

*Sanders, R. W., "Communications Efficiency Comparison of Several Communications Systems", PROC. IRE, Vol 48, pp. 575-588, April 1960.

It is considered very important that some positive external pressure gradient be maintained to prevent efflux of electrolyte under space conditions. Batteries normally may be completely sealed <u>only</u> under very special conditions which will prevent rupturing from internal gas pressure. The recommended procedure is for the entire battery-pack container to be pressurized with a relief valve for the container set at about 5 psi. An alternate procedure is to fit each individual cell with a pressure relief valve. Manufacturing drawings for such a valve are available from AFSWC (SWTTS).

The general design should insure that batteries in flight do not reach temperatures near $0^{\circ}C$. The overall ampere-hour capacity is greatly reduced at low temperatures.

All power-consuming devices should be designed to accept voltages higher than nominal, since the discharge characteristics of silver-zinc cells normally start at a high "peroxide" voltage before reaching a relatively constant plateau voltage.

Internal payload battery power will normally be conserved after flight batteries are installed by the use of an external power source and umbilical controls which switch the payload from internal to external power. Operation on internal power can usually be limited to 15 minutes before flight, including all compatibility checks, F-1 Day, and pre-launch checks.

A set of recommended battery procedures, "Yardney Silvercel Battery Service Instructions", is available from AFSWC (SWTTS) upon request.

D. <u>Available Volume-Heatshield Restraints</u>.

The total volume available for the payload is shown in Figure 5, which shows the I-section interstaging and the fourth-stage spherical motor. At least one inch clearance from all surfaces must be maintained. This clearance, plus an allowance for a typical payload adapter results in a usable volume as shown in Figure 6 of about 4 cubic feet. A typical payload adapter section, made of plastic foam molded to fit the fourth-stage motor is described in Section III.D. A radar corner reflector assembly, as shown in Figure 8, mounts at the base of the fourth-stage motor by means of a high temperature adhesive. Total weight of this unit is 8 ounces. For a full size scale drawing of the nose cone interior and payload area, request AFSWC, Test Directorate Design Branch Drawing E-502, "Payload Area Coordination Layout", from AFSWC (SWTTS).

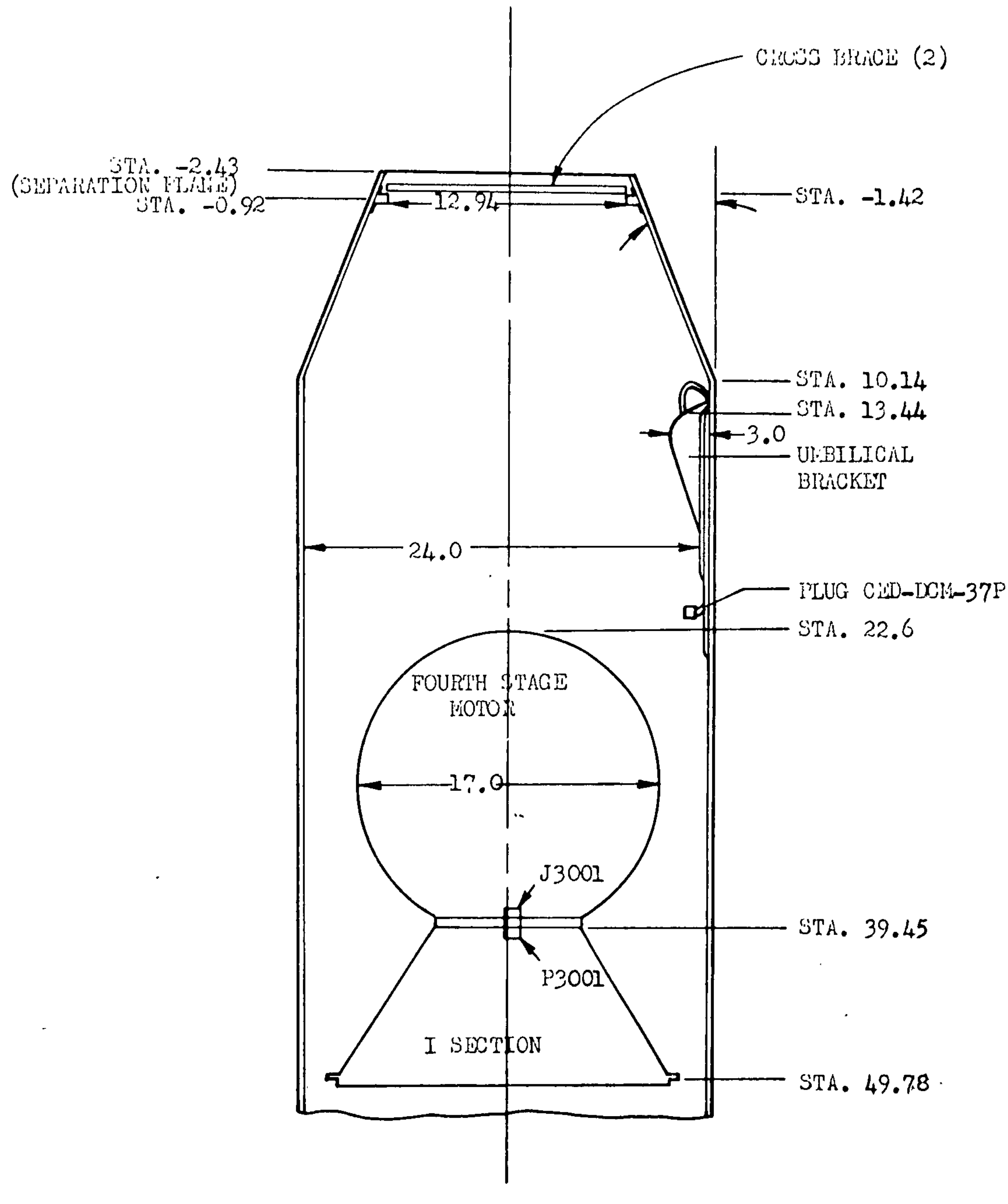
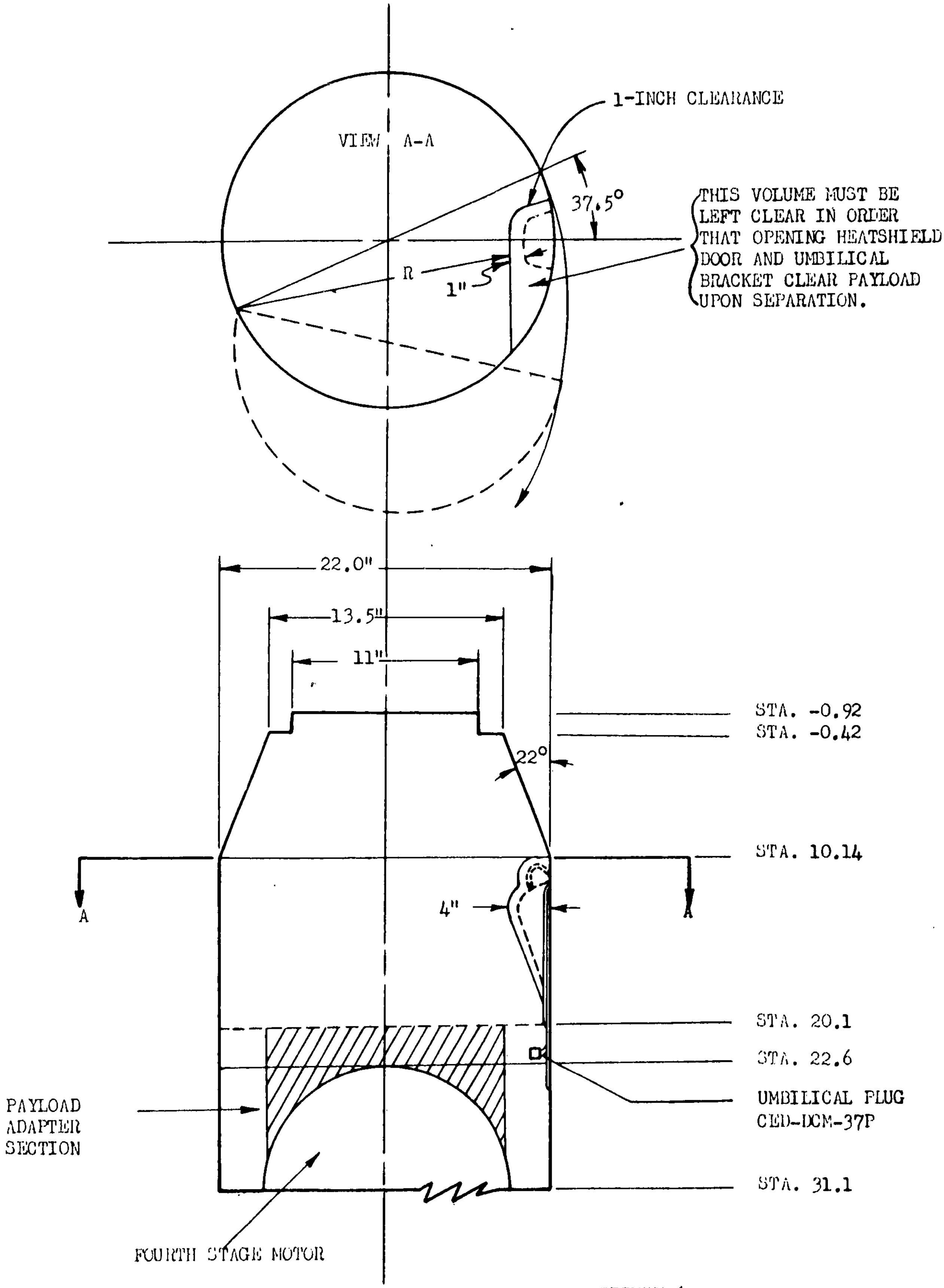

CROSS BRACE (2)
STA. -2.43
(SEPARATION PLANE)
STA. -0.92
12.94
STA. -1.42
STA. 10.14
STA. 13.44
3.0
UMBILICAL
BRACKET
24.0
PLUG CED-DCM-37P
STA. 22.6
FOURTH STAGE
MOTOR
17.0
J3001
STA. 39.45
P3001
I SECTION
STA. 49.78

FIGURE 5.



VIEW A-A
1-INCH CLEARANCE
37.5°
THIS VOLUME MUST BE LEFT CLEAR IN ORDER THAT OPENING HEATSHIELD DOOR AND UMBILICAL BRACKET CLEAR PAYLOAD UPON SEPARATION.
R
1"
22.0"
13.5"
11"
STA. -0.92
STA. -0.42
22°
STA. 10.14
4"
A
A
STA. 20.1
STA. 22.6
UMBILICAL PLUG CED-DCM-37P
PAYLOAD ADAPTER SECTION
STA. 31.1
FOURTH STAGE MOTOR

FIGURE 6.

Figure 7.

CORNER-REFLECTOR ASSEMBLY

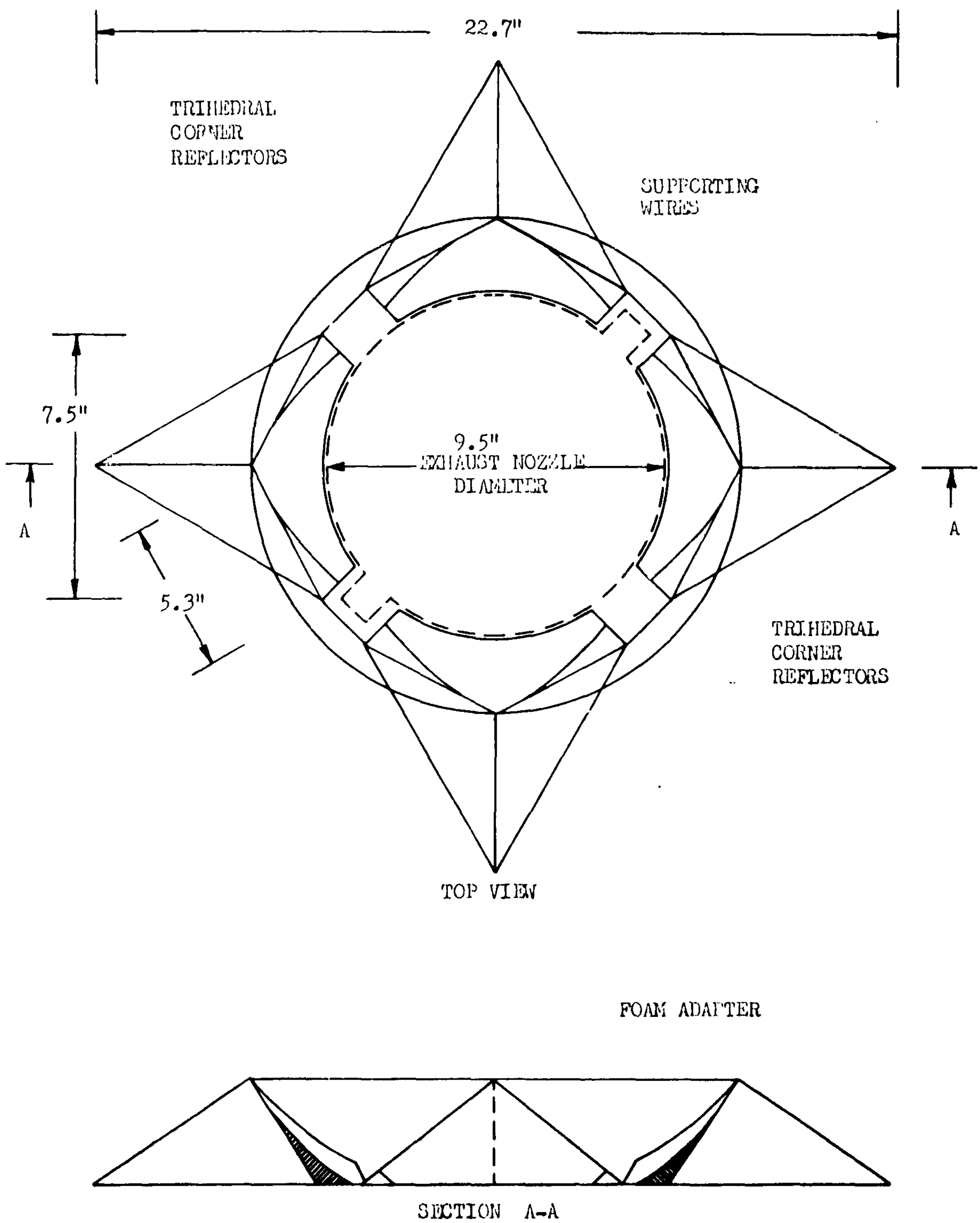


FIGURE 8.

E. Environmental Tests.

The determination of an adequate test program is a prime problem in environmental testing for an effective space program. On a statistical basis, it has been shown that if the payload is to accomplish its mission, the reliability of each component in a large system must be very high. The approach taken to achieve this reliability is twofold, consisting of Design Approval Testing and Flight Acceptance Testing.*

Design Approval Test: In this test the various stages of payload development through the prototype unit (particularly equipment of new design) are required to withstand conditions approximately 50% more severe than those anticipated in the flight environment. Payload units which have been subjected to Design Approval Tests are generally not acceptable as flight units because of the severity of the tests imposed.

Flight Acceptance Testing: Even though the prototype unit has passed the Design Approval Test, it is still necessary to conduct simulated environmental tests on the final flight unit. In general, the Flight Acceptance Test will be approximately equal to the expected flight environment.

The following minimal flight acceptance environmental tests are established by AFSWC (SWTTS). Experimenters are expected to perform any additional tests deemed necessary to prove the payload flight-worthy. A report summary of each test performed will be forwarded to AFSWC (SWTTS). A sample form for this report is shown in Figure 11. AFSWC (SWTTS) will furnish SSD with a certification that the payload is dimensionally, functionally, and environmentally compatible with the SLV-1B vehicle and the expected environment.

Complete environmental testing cannot be overemphasized. It is also deemed very important that the payload be operating (in the configuration it will be in during the boost portion of the flight) during as many of the environmental tests as possible. This is of great value in locating such possible troubles as shorts occurring under acceleration and vibration.

*"Considerations Affecting Satellite and Space Probe Research...", NASA Technical Report R-97, 1961.

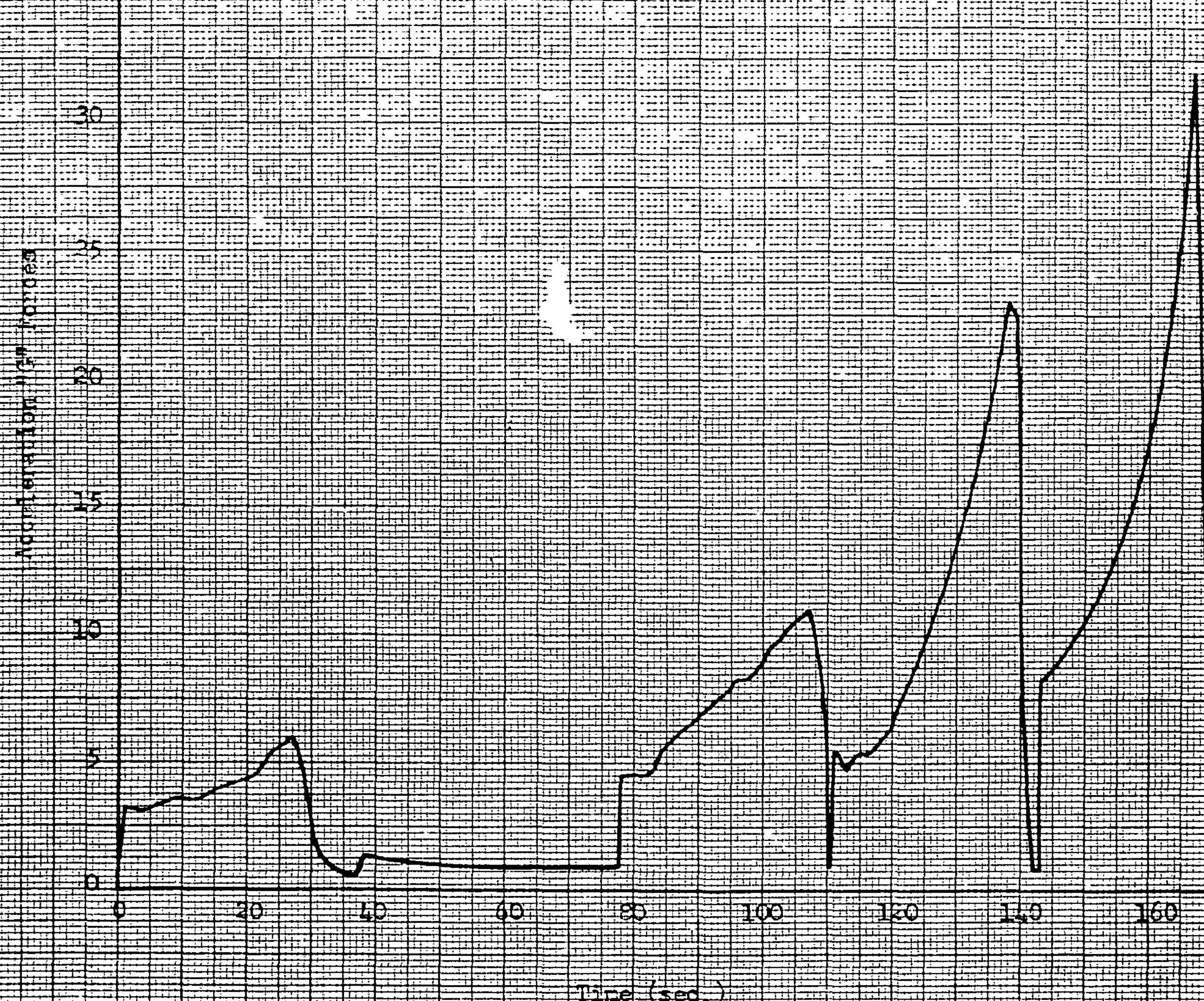
1. Shock. Principal shock environment, other than normal handling, shipping, etc., will be longitudinal (i.e., along the vehicle thrust line) and will consist of igniter acceleration peaks. Accomplishment of one test to a half sine pulse of 30g's peak amplitude and of 10-15 milliseconds duration in thrust axis only is sufficient for this environment.

2. Acceleration. Maximum longitudinal acceleration for the third stage is about 27g and for the fourth stage is about 35g (See Figure 9). One acceleration test along the thrust axis to the maximum calculated level for the specific payload weight during fourth-stage burning for a period of three minutes is sufficient for this test. For lateral acceleration, the maximum spin rate is 3.2 revolutions per second. Lateral g's may be computed using this rate and location of the mass from the center line. Spin-up from 0 to 3.2 rps occurs in a time period of 0.10 second. A single spin test that satisfies these conditions is acceptable.

3. Vibration. The second-stage motor (X-259) is the "roughest" burning of all stages. Vibrations of up to 13g's have been measured at the second-stage motor, principally in the frequency range above 800 cps. All frequencies appear to be present. It is expected that total vibration seen by the payload from aerodynamic buffeting and motor burning does not exceed 10g's, with components present being in the range of frequencies below 250 cps. The longitudinal vibration test will consist of vibration in the longitudinal plane of 0.01 double amplitude (inch) in the range of 20-65 cps and 10g's in range of 65-2000 cps. This will be a 3-minute excursion from 20 cps to 2000 cps and return to 20 cps. The lateral vibration test can be eliminated if the completed payload is tested while mounted to the payload adapter section. In this configuration, the longitudinal test induces the expected flight lateral vibrations by cross coupling. Subsystems, components, and payload without the payload adapter will be vibrated in the lateral plane, the reference being the mounting position on the payload, at 0.01 double amplitude (inch) in the range of 20-65 cps and 2g's in the range of 65-2000 cps. This will be a 3-minute excursion from 20 cps to 2000 cps and return to 20 cps. Those subsystems and components requiring structural support from the payload adapter may be removed from the structure in this test and tested individually.

4. Temperature. Normal rise of temperature within the nose cone due to aerodynamic heating will be about 30°F. This rise will be from sea level temperature of about 70°F, unless some pad payload temperature control is employed. Temperatures theoretically as high as

Figure 9.


Acceleration in "g" forces
0
5
10
15
20
25
30
35
0
20
40
60
80
100
120
140
160
Time (sec.)

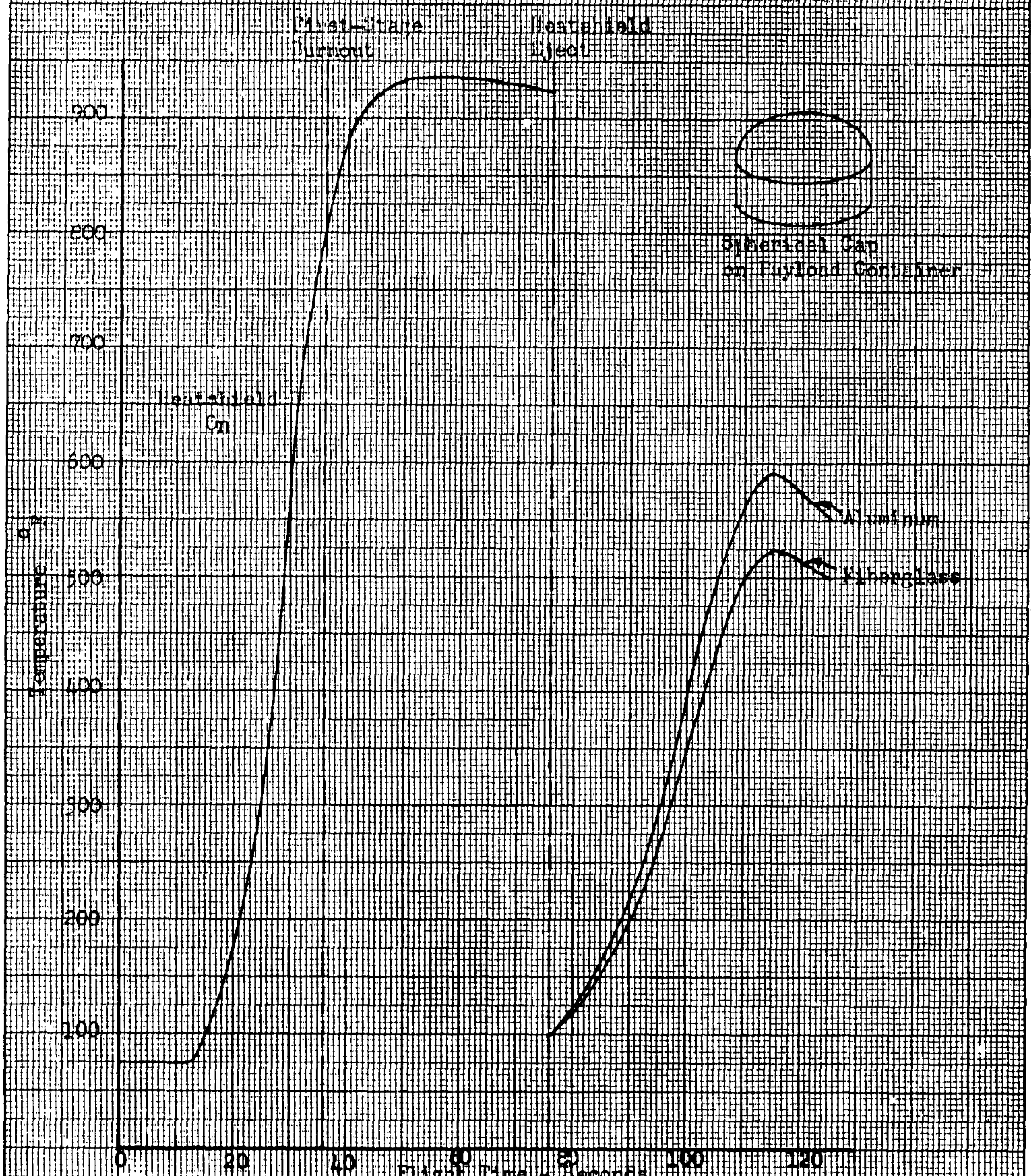
600°F stagnation skin temperature for 10 seconds may be expected when the nose cone ejects. Figure 10 is the stagnation temperature versus time family of curves for the aerodynamic heating on the payload after nose cone ejection. Stagnation temperature effects may be minimized by proper design (consulting service in this area is available from AFSWC/SWTTS). The temperature test will consist of a heat shock test and a temperature soak test. The heat shock test is highly recommended because of the high stagnation temperature. The program for this test can be taken directly from Figure 10. The soak test will consist of soaking the payload at the user's designated stabilization temperature for a minimum time duration of 24 hours. Functional tests and visual checks will be performed in the same manner as in the humidity test. In addition, successful operation at the design temperature extremes shall be demonstrated.

5. Humidity. Humidity conditions will be the same as those present at the launch site unless controlled in some fashion. The nose cone does not provide an air-tight or moisture-tight covering and the completely assembled vehicle can be expected to be on the pad as long as 4 days before launch. The minimum requirements of this test will consist of soaking the payload in 95% relative humidity, 77°F for a period of at least two days. Since this test simulates conditions on the launch pad, it is not necessary to operate the payload full time. A minimum of four functional tests will be performed at regular time intervals such that the last test occurs prior to conclusion of the test. A visual inspection of accessible components and circuitry will be conducted and results noted in the test report. This test can be eliminated if a form of pad humidity control is to be employed (such as dry nitrogen purging).

6. Altitude. The required altitude testing is directed primarily toward assuring successful telemetering system performance; i.e., assuring that flashover does not occur (on those systems employing high voltage components) within transmitters, and that corona breakdown of antennas does not occur. Similar applications within the payload experiment should also be tested. A flight-length test is specified in order that hermetically sealed units can be given a realistic check.

This test consists of exposing the payload to vacuum conditions of 130,000 feet for a period of time equal to the anticipated flight time. Functional tests will be performed in the same manner as in the humidity test. An altitude of 130,000 feet (2.36 mm Hg) must be maintained since this is the most likely flash-over pressure.

Figure 10.



Stagnation Point Skin Temperature

Date Performed_____

Report of Test Performed on Payload __________

Test:

Limits and Duration:

Results (Diagrams, curves, comments; attach additional pages as required):

Signature:____________________ Page 1 of ___
(Project Engineer)

Figure 11

If the system successfuly completes this test, additional tests at minimum achievable chamber pressure are recommended. Testing of materials in a space environment (sublimation, etc.) should be at a pressure of 10^{-6} in. Hg or less. Exposure to pressure lower than 10^{-10} mm Hg may be expected in flight.

See also Section II.K.3, "High Voltage Turn-On", for particular altitude considerations.

7. Dynamic Pressure. At heatshield separation, a dynamic pressure loading of 5.33 pounds per square foot may be expected. No environmental test for this condition is prescribed.

8. Dynamic Balance. The payload, together with payload adapter section, will be dynamically balanced to within one inch-ounce, if possible. This is not necessarily an environmental test since this is a necessary operation before the payload can be mounted to the vehicle. If the spin rate is 3.2 rps, the lateral acceleration g forces applied simulate those to be encountered in flight.

9. Moments of Inertia. Measure I_L and I_T, the longitudinal and transverse moments of inertia. Moments of inertia about the yaw axis I_Y and about the pitch axis I_P should be equal.

Many in-house environmental testing facilities exist within the Federal establishment. Assistance in locating and/or arranging for the use of these facilities will be provided upon request to AFSWC (SWTTS).

F. Attitude Stabilization.

Payload attitude stabilization is normally obtained by spinning. At fourth-stage burnout, a spin rate on the order of 3 revolutions per second is normally present. This attitude, which can be theoretically obtained by calculating the burnout vector, is normally maintained throughout the remainder of the free-fall flight if the ratio of moment of inertia taken about the longitudinal axis to the moment of inertia about the transverse axis is greater than one.*

*Grasshoff, L. H., "Influence of Gravity Upon Satellite Spin Axis Attitude", ARS Journal, Vol 30, No. 12, Dec 1960.
Manger, W. V., Journal of Geophysical Research, Vol 65, No. 9, pp.2992.

$\frac{I_L}{I_T} > 1$ The period of stability is also dependent upon the spin rate. If the payload remains attached to the fourth-stage motor casing, this, of course, must also be considered. Inherent stability may be increased by keeping the overall height of the payload to a minimum and by placing heavy components, such as batteries, as far from the axis of rotation as possible.

Dynamic balancing of the payload is also required from long-term payload stability considerations and from a vehicle structural standpoint. Judicious placement of components to obtain a natural configuration near dynamic balance will pay weight dividends during the final dynamic balance, since less dead weight will have to be added to achieve balance.

Estimated moments of inertia about the roll, yaw, and pitch axes (for the payload and payload adapter only), c.g. location, and total payload weight may be requested by the vehicle contractor 6 to 8 months prior to launch. It is recommended that I_L and I_T for the completed payload actually be measured and recorded.

Attitude stabilization is normally required in order to achieve payload temperature control. See the discussion in Section G. Temperature Control.

G. Temperature Control.

Severe weight limitations normally limit the method used for SLV-1B spacecraft temperature control to passive techniques.

The temperature of a satellite or space probe depends only on radiative heat transfer. The stabilization temperature is determined by the amount of radiative heat which the body receives, the heat generated internally by the payload components, and the heat which the body reradiates or reflects to the surrounding space. The time rate of change of payload temperature can be expressed by the following equation.*

$$\frac{dT_s}{dt} = \frac{\alpha_1(I_{s_1} + I_{s_2}) + \alpha_2 I_E + P - \varepsilon\sigma T_E^4 A}{MC}$$

*"Considerations Affecting Satellite and Space Probe Research...", NASA Technical Report R-97, 1961, page 49.

where T_s = temperature of spacecraft shell.

t = time

α_1 = absorptance of vehicle's surface to solar radiation.

I_{s_1} = incident solar power.

I_{s_2} = solar power reflected to vehicle from earth.

α_2 = absorptance of vehicle's surface to terrestial radiation.

I_E = terrestial power incident upon the vehicle.

P = power dissipated within the vehicle.

ε = emittance of spacecraft's surface.

σ = Stefan-Boltzmann Constant.

T_E = effective temperature of the earth.

A = vehicle's surface area.

MC = vehicle's heat capacity.

Through the use of specially developed organic coatings with a specific absorptivity (α) and emissivity (ε), a desired heat flux balance can be maintained; this results in the desired stabilized temperature if the spacecraft attitude, with respect to the sun, remains constant. AFSWC (SWTTS) has performed the heat analysis and ordered the required organic coatings* for experimenters in the past, so this service is available if desired. To perform this analysis, the following information is needed:

1. The orientation required by the experiment (if any) of the longitudinal axis of the payload with respect to the sun.

*Spacecraft Temperature Control, Space/Aeronautics, July 1961.

2. The exterior geometry of the payload.

3. Amount and location of internally generated heat.

4. Materials used in the spacecraft structure (for determination of conductivity, etc.).

5. Stabilization temperature desired.

A launch window will be determined by these considerations; i.e. a constant attitude with respect to the sun.

In general, care should be taken to thermally link all components which require heat transfer (such as transmitters requiring heat-sink and batteries which require protection from low temperatures) by a good heat conducting material. Thermal isolation of components is to be discouraged. In this connection, certain magnesium alloys are poor heat conductors and should therefore not be used in applications where heat flow is desired.

Design temperature operating limits are commonly 0°C and 65°C. Stabilization design temperature is nominally 25° to 30°C.

H. Vehicle Performance Instrumentation.

The following booster performance instrumentation is required in all payloads:

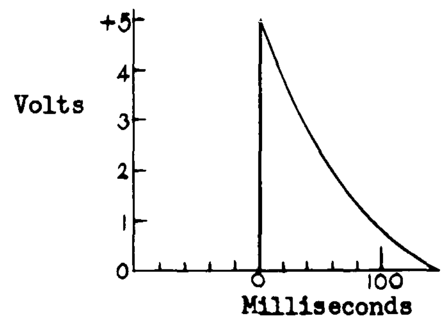
1. One longitudinal accelerometer.

2. One lateral accelerometer.

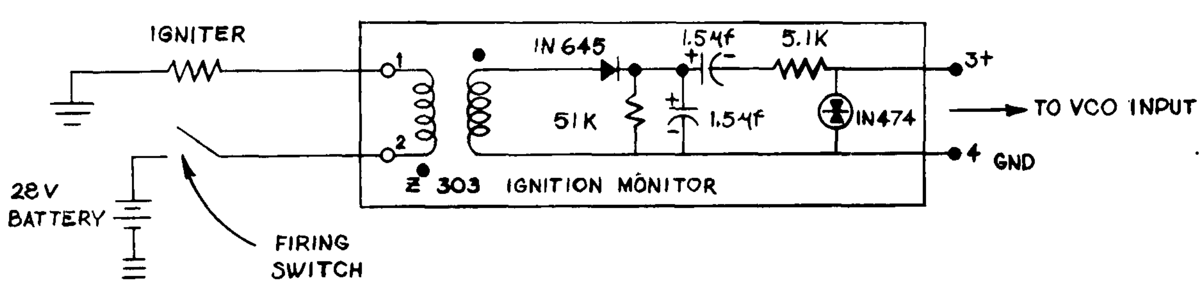
3. Ignition monitors for Stages 2, 3, and 4.

This is minimum instrumentation. Additional instrumentation will be included if possible (i.e., if weight, volume, and data link permit). This additional instrumentation may include such items as pressure transducers, magnetic aspect sensor, additional lateral accelerometer vibration pickup, and thermistors.

This instrumentation must be integrated into the payload and payload telemetry system. Normal procedure is to utilize a changeover device (such as a time delay relay), which will deactivate booster instrumentation and switch telemeter channel inputs from booster instrumentation to experiment sensors shortly after fourth stage burn-out. AFSWC (SWTTS) will normally procure suitable accelerometers, magnetic aspect sensors, time delay relays, pressure transducers, and vibration pickups, utilizing funds provided by the program office. In the event that experimenters cannot utilize the transducers provided and desire that AFSWC (SWTTS) initiate procurement action for a different type, full information, specifications, and sole source justification for the desired type must be provided by the experimenter. Ignition monitors are incorporated into the vehicle by the vehicle contractor. Two pulses for each stage ignition are supplied to the payload through wiring from I section. (See Section III.B. for details on this wiring.) These pulses may be impressed upon the output of another transducer, as long as they are distinguishable. The pulse supplied is of the following form:



Ignition Monitor Schematic:



Proposed vehicle instrumentation for each payload should be coordinated with AFSWC (SWTTS) well in advance of design freeze.

I. Antenna System

An accurate antenna pattern must be submitted with the first PR submission (See Flight Planning Documentation for explanation of this term). This pattern, together with particulars on the telemetry system, allows the Range to assess the tracking-data-gathering problem and to decide if the desired support can be provided. Patterns should be prepared to conform to IRIG Document 102-61, Standard Coordinate System and Data Format for Antenna Patterns.

Primary requirements for a SLV-1B antenna system are:

1. Uniform pattern with no nulls greater than 3 db in the aft 90° conical section.

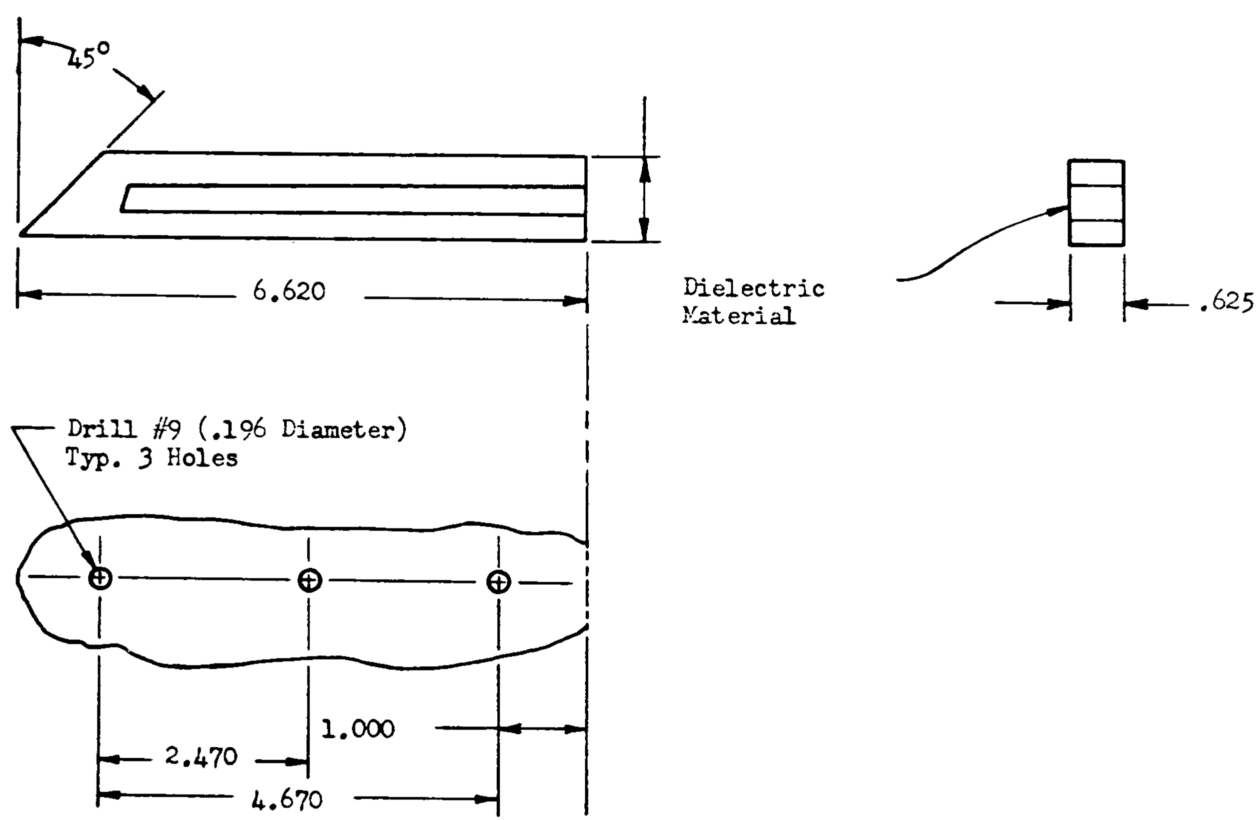
2. Reliability of connections, matching, continued use in the event of collision.

3. Power handling capability (no corona when tested under full power at 130,000 feet altitude).

4. Light weight.

Two antenna systems used successfully in SLV-1B vehicles are described on the following pages. The antenna shown in Figure 12 was designed by Physical Sciences Laboratory, New Mexico State University, Las Cruces, New Mexico. The system consists essentially of two symetrically located "quadraloop" elements fed 180° out of phase or in phase, depending upon the pattern desired. The pattern is very uniform, is linearly polarized, but may be received with either left or right-hand circularly polarized antennas because the E and H fields are orthogonal. This system is rugged and reliable. Antennas which weigh about four ounces each without phasing cables have a power-handling capacity of about 2 watts each. Special precautions must be taken to protect the dielectric material from moisture. One of these antennas is visible in the photograph of Figure 7.

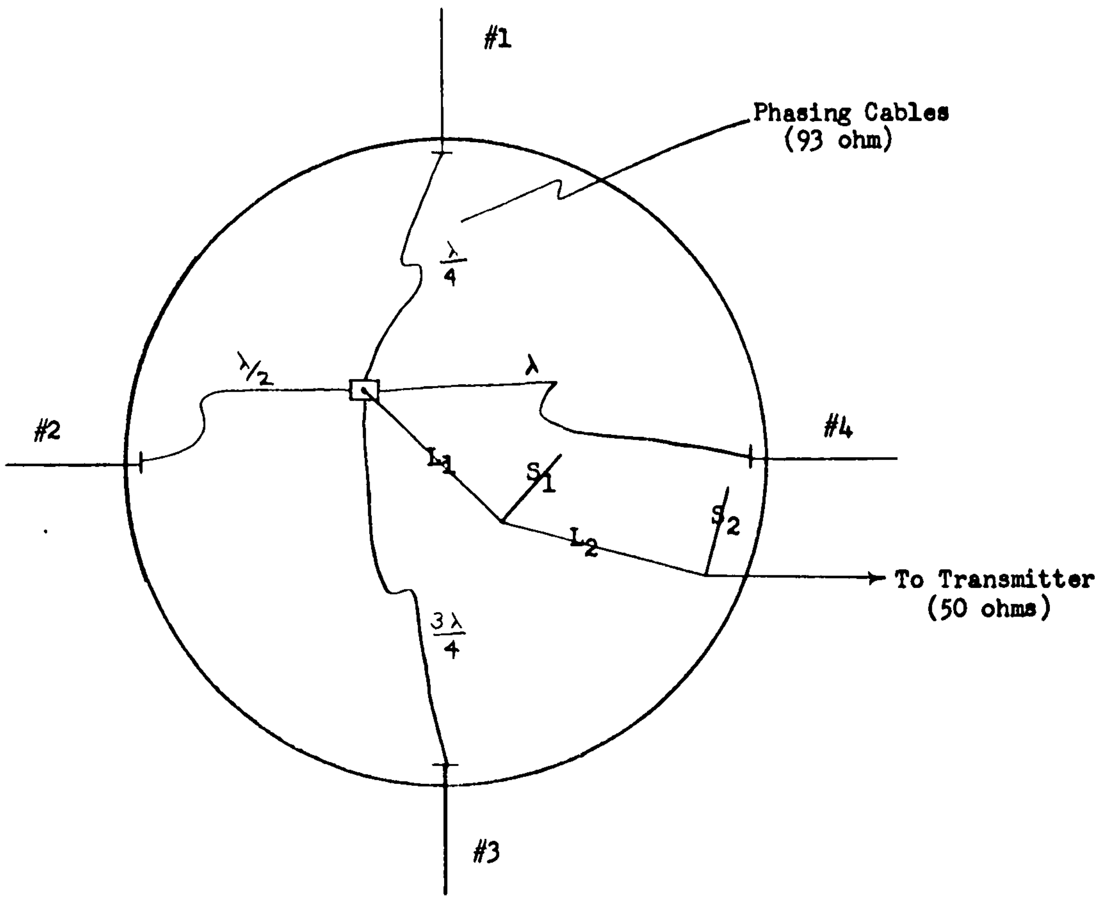
The antenna system shown in Figure 13 was designed by Division 1424-3, Sandia Corporation, Albuquerque, New Mexico. It is essentially a crossed element phased array (See VHF Techniques, Vol. I, p. 132, First Edition). Polarization is circular with an axis of symmetry along the longitudinal axis of the missile. Two patterns are utilized:



ANTENNA MOUNTING TEMPLATE

FIGURE 12.

Typical New Mexico State University Quadraloop Antenna Element
NMSU Dwg No. PSL 1710-4



Bottom View

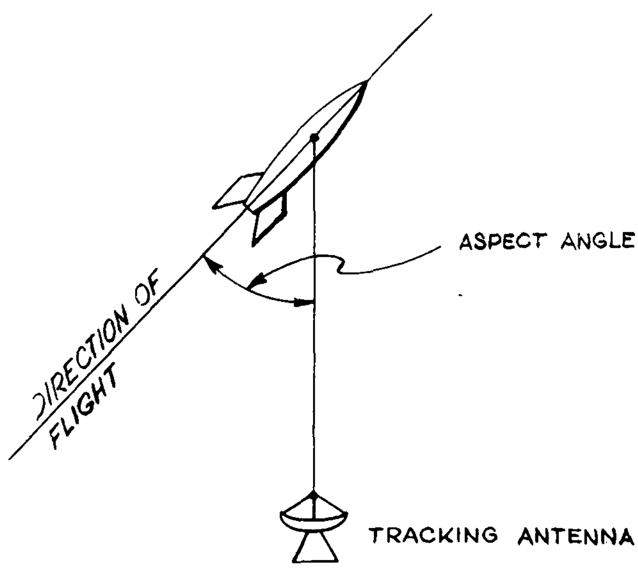
Polarization is Right-hand Circular

Figure 13.

(a) with the heatshield ON and the antennas folded upward constrained by the heatshield; (b) with the nose cone OFF and antennas extended laterally, normal to the longitudinal (roll) axis of the missile. The folding feature is necessary to allow inclusion of this turnstile-type antenna within the heatshield dimensions. (Refer to Figure 13.) The phasing cables provide the necessary 90° phase difference between elements. L_1 and L_2 are adjusted, in conjunction with matching stubs S_1 and S_2, to provide a match in the "up" position. At heatshield separation, stub S_1 is removed and the system is again matched in the normal extended position. This system has an acceptable pattern, has power handling capacity on the order of 20 watts, and is exceptionally light (.5 lb for all components). Its disadvantages are in the "folding" feature; i.e., the multiple match required, and the possibility of damage to the system upon heatshield separation.

In general, antenna patterns, tuning, and matching procedures vary markedly with payload configuration. For this reason, an accurate mockup of the payload (and fourth-stage motor casing) must be available to personnel running patterns, and matching antennas and cabling. See Section III.A., "Payload Orientation with Respect to Vehicle", for recommended antenna orientation within the heatshield.

As an aid to assessing antenna pattern suitability, all trajectory printouts supplied to experimenters by AFSWC (SWTTS) will include aspect angle. This is the angle made by the line from a particular tracking station with the spacecraft burnout vector. An aspect angle of 0° indicates that the tracking station is looking at the tail of the missile.



J. Umbilical Connections.

Thirty-seven umbilical connectors to the payload are available. These consist of insulated conductors numbers 1 through 24, shielded conductors numbers 25 through 36, and number 37, which is a ground common to blockhouse earth ground. All conductors are #20 guage wire, whose maximum allowable current carrying capacity is 15 amperes. A Cannon CED-DCM-37S connector on the payload mates with a Cannon CED-DCM-37P connector, which remains with the heatshield upon its separation. See Section III.A., "Payload Orientation with Respect to Vehicle", for orientation of the umbilical connection within the heatshield.

Some normal payload blockhouse umbilical functions are:

1. Switching from external power to internal power.

2. Switching from internal power to external power.

3. Supply of external power.

4. Switching of control relays from booster performance instrumentation to experiment and reverse.

5. Monitoring safe-arm relay positions for ordnance items in the payload.

It is imperative that NO voltages from the internal system be present on any of the umbilical conductors at lift-off. The design of this system should be such that all payload umbilical conductors at the DCM-37S connector, when in flight, may be considered shorted WITHOUT affecting payload operation. The use of magnetic latching-type relays or Ledex-type rotary stepping switches in internal umbilical-controlled switching functions is therefore advised.

Umbilical separation occurs at launch by physical forward movement of the vehicle. Separation of the interior umbilical (from payload to heatshield) occurs when the heatshield separates just prior to second-stage ignition.

K. Special Considerations.

1. Separation from Fourth Stage Motor Casing. Normally, the body boosted into a ballistic trajectory will consist of the payload, the payload adapter, and the burned out steel casing of the fourth-stage motor. In some instances (as in magnetometer experiments, for example), the presence of the steel motor casing in the vicinity of the experiment cannot be tolerated. In such a situation, a means for detaching the payload from the motor, and separating the two by some distance must be devised.

2. Pre-launch Payload Purging. No gas or liquid line is presently incorporated into the umbilical system. If an experiment requires continuous fluid or gas flow into the payload (such as dry nitrogen purging), this capability must be designed into the missile-payload compartment-umbilical release system. The usual approach is to utilize flexible tubing to the payload in conjunction with access provided by the umbilical doors (See AFSWC, Test Directorate Design Branch Drawing E-502, "Payload Area Coordination Layout"). Purging requirements must be shown in the PR, together with the expected amount of dry nitrogen to be used.

3. High Voltage Turn-on. Experience has shown that experiments containing voltages of 400 volts or greater often experience break-down (corona) at extremely high altitudes, even though at this altitude, voltage breakdown should theoretically not occur. This is very often a consequence of material out-gassing. If the experiment contains an exposed high voltage source, low pressure vacuum chamber tests with the entire payload should be made to establish the time required for out-gassing to reach a level satisfactory for high voltage turn-on. This time, rather than a certain altitude, should be used in the flight programmer to initiate high voltage turn-on.

4. RF Interference. It has been found that RF interference with low-level amplifying electronics in the payload may be experienced if special precautions are not observed. Because of the proximity of antennas to the payload, the telemetering transmitter generates high RF field intensities in the area of the payload. The best protection against such interference is: (a) shield electronics thoroughly; (b) install a tight-fitting RF shield over the entire assembly (this shield may also serve to achieve payload temperature stabilization and reduces the effects of stagnation temperatures at heatshield separation); (c) do not allow wiring, which might carry RF into electronics to be exposed;

(d) decouple RF where possible; and (e) if the payload will remain attached to the payload adapter section during flight, coat the payload adapter interior with conducting material such as silver paint or copper plating so that the payload underside is essentially inclosed.

III. PAYLOAD FABRICATION & ASSEMBLY.

A. Payload Orientation with Respect to Vehicle Heatshield Operation.

Figure 14 is a view of the vehicle looking aft. On the launcher, the vehicle is suspended along the 0° line (launcher side of the yaw axis). The payload (forward) umbilical also enters the vehicle along this line. Downrange is on the 180° side of the vehicle. As the vehicle clears the launcher, spin rockets fire, rotating it counterclockwise as shown. The heatshield consists of two halves, the separation plane of which is along the 37.5° - 217.5° line. The seams contain metal reinforcing strips (the remainder is .6" fiberglass) and it is therefore recommended that antennas be located as far from these strips as possible:

1. For a two-element system, antennas should be at 127.5° and 307.5°.

2. For a four-element system, antennas should be at 82.5°, 172.5°, 262.5° and 352.5°.

At heatshield separation, the right half section rotates clockwise about the seam point at 217.5°. The left half section (shown in Figure 14) rotates clockwise about the seam point at 37.5°. After a rotation angle of 26°, the hinges are mechanically unlatched and the heatshield halves are flung outward by centrifugal force. The payload-umbilical connector, CED-DCM-37S, mates with the DCM-37P "vehicle" connector along the separation plane as shown, such that the pull exerted on the payload connector by the separating heatshield section is normal to the plug. This mating may be at any vehicle station between 13.44 and 39.45.

B. Wiring to Payload from I-Section.

Plug P3001 (CED-DM-9P), carrying wiring from ignition monitors as shown below, mounts on the I-section and mates with jack J3001 at the base of the fourth-stage motor (Station 39.45). P3000 and J3000 also mate 180° from P-J3001, but are a dummy connection for balance only. The azimuth of these plugs, in respect to the degree (umbilical) azimuth is unknown until fourth stage and I-section are mated at the launch site in accordance with the following instructions:

1. Engage threads of diaphragm to the XM-85 motor until diaphragm bottoms out on internal shoulder of XM-85 motor.

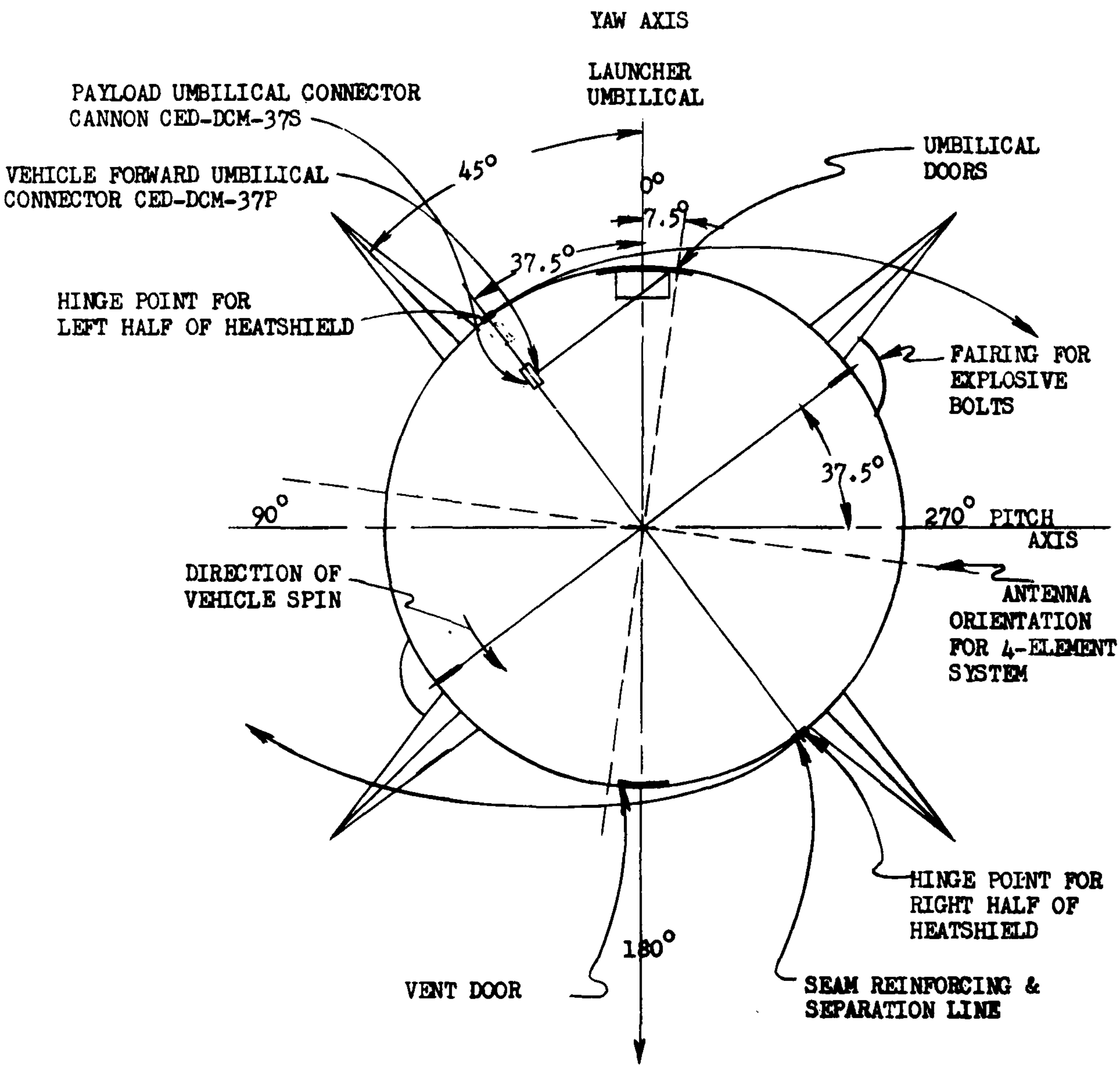


Figure 14

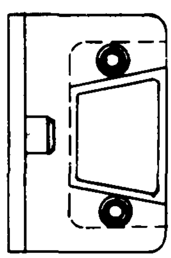
VIEW LOOKING AFT (From VA Dwg 322-00040)

2. Engage threads of I-section onto the protruding threads of the diaphragm until I-section bottoms out on XM-85.

3. With steps 1 and 2 completed, the orientation of the XM-85 motor with I-section (and, hence, with the entire vehicle) has been established. Scribe lines for appropriate drilling and tapping for attaching plugs P3000 and P3001 to I-section.

4. Leads from payload to J3001 to be soldered in field.

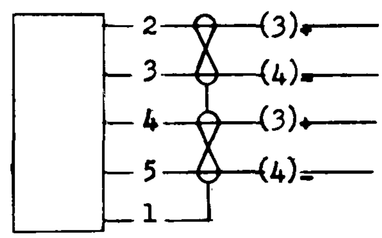
At fourth-stage ignition, these connectors (and fourth stage and I-section) separate along the station 39.45 line. Jack 3001 can be expected to experience <u>severe heating</u> and <u>likely shorting</u> during fourth-stage motor burning. It is, therefore, important that such a shorting not affect the payload operation in any manner (A-C couple).



TOP VIEW
J3001 & BRACKET.
CED-DCM-9S
(Mounting on XM-85
Motor)

To
J3001
and Pay-
load.

P3001 (CED-DCM-9P Mounts on I-SECTION)

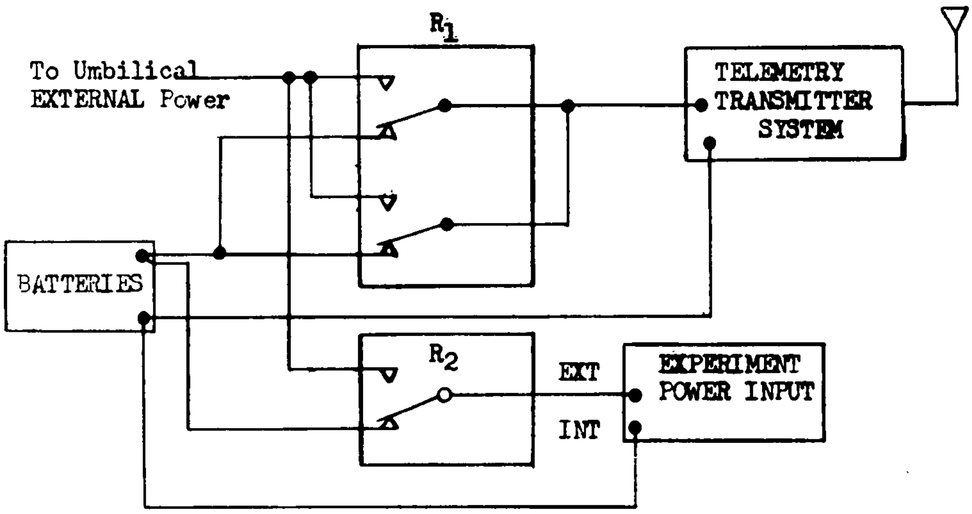


From Ignition
Monitor Z303
System 1

From Ignition
Monitor Z303
System 2

C. Power Distribution Wiring.

Since the telemetry carrier signal provides the only means of tracking the payload at extended ranges, it is extremely important that all precautions be taken to insure that the telemetry transmitter continues to operate, despite other payload malfunctions which might occur. Such a philosophy should be applied throughout the payload and is illustrated here by a recommended power distribution system:



Relays R_1 and R_2 are simultaneously controlled by an external-internal power command. The wiring is such that, if a complete in-flight short is experienced in the experiment thereby burning the contacts of R_2 open, R_1 and the transmitter operation are unaffected. R_2 serves, in effect, as a fuze in the power line to the experiment. R_1 and R_2 should be separate relays. Note that wiring for the internal power systems is commoned only at the battery terminals. Under no circumstances should the internal power to the telemetry system be commoned with power to other systems except at the battery terminals. The power distribution system should be created in mockup, utilizing batteries, proper size and length of wire, relays, etc., and a short circuit test conducted to insure that the telemetry system remains in operation.

D. Recommended Payload Adapter Arrangement.

1. It is the responsibility of the payload-building agency to also supply the payload adapter section; i.e, that structure which mounts to the fourth-stage motor and supports the payload (spacecraft) structure. The payload adapter section is governed by the following restraints:

a. It must not be welded, bolted, or otherwise secured to the fourth-stage motor casing in any manner which requires that machine work be performed on the casing. The casing is .030-inch thickness steel and is not available for machine work after being filled with propellant. No stress concentrations (such as would occur from welding) are allowable in the casing.

b. The loading presented to the fourth-stage motor surface must be uniformly distributed over the forward hemispherical area. Points of load concentration must be avoided (a maximum of 3 psi loading under no-g conditions).

c. Since the payload adapter section is essentially part of the payload, the weight restriction on the adapter is severe; i.e., the weight of the adapter section must be taken as part of the maximum allowable payload weight.

2. The payload adapter section, utilized successfully on previous SLV-1B flights, satisfies the foregoing requirements. It consists of a plastic foam section, molded to fit the hemispherical motor surface on the underside and flat on the upper surface. It is bonded with a high temperature adhesive (Shell Epon V1) to the motor casing. This foam has the following specifications:

a. Density: 2.5 lb per cubic foot.

b. Room temperature compressive strength: 40 psi

c. Room temperature tensile strength: 40 psi

d. Maximum continuous strength: 40 psi

Additional characteristics are:

e. Volume of payload adapter section: 0.7 cubic feet.

f. Weight of payload adapter section: 1.6 to 1.8 lbs.

g. Surface area, motor-to-adapter section: 300 in^2 .

h. Surface area, payload to adapter section: 225 in^2.

i. Weight of adhesive, adapter section to motor: .4 to .5 lbs.

j. Bonding operation: 12 hours.

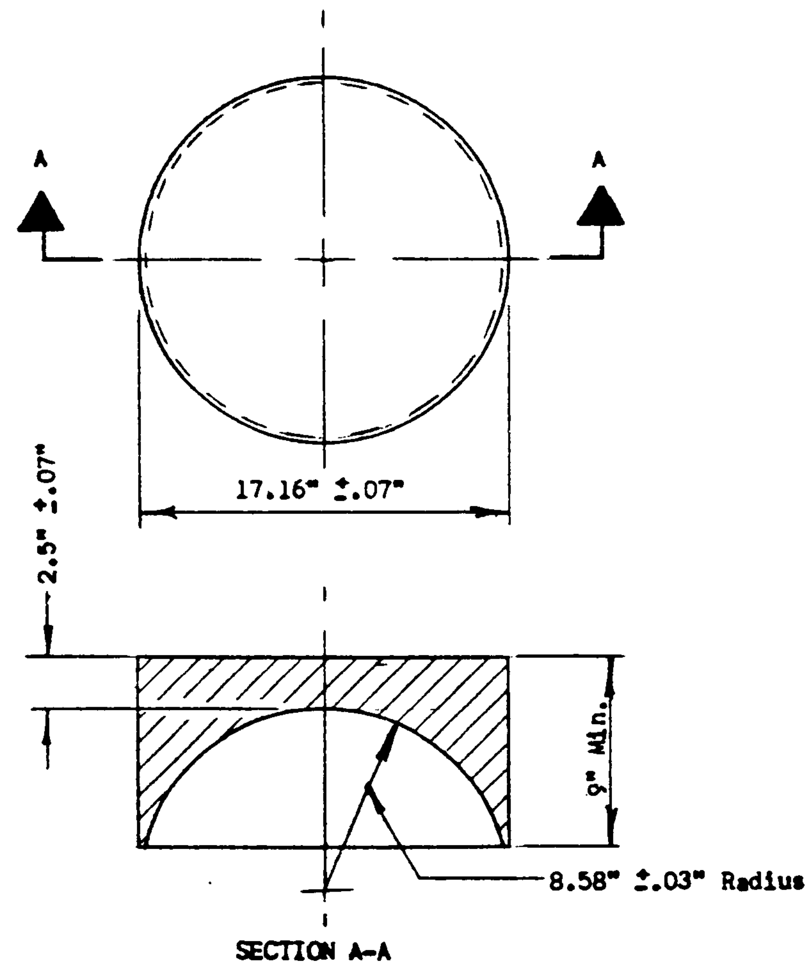
A suggested source for a suitable plastic is: Lone Star Plastics
P.O. Box 12007
Fort Worth, Texas

The drawing on the following page may be used in procurement specifications to give proper dimensions and tolerances.

Normally, the bonding operation occurs at the launch site and is performed by the launch agency. The adhesive will be supplied by AFSWC (SWTTS). A publication, "Procedures for Bonding Plastic Foam Payload Adapter to Fourth-Stage Spherical Motor - SLV-1B", is available from AFSWC (SWTTS) upon request.

It has been found that strength capabilities of the foam can be greatly increased, if deemed necessary, by covering all surfaces with one layer of fiberglass cloth, adhered to the foam, utilizing a standard fiberglass-resin technique. The weight penalty of this process is on the order of one pound.

The plastic foam has been found to have good vibration damping qualities and thermal qualities (conductivity) roughly comparable to asbestos. Its principal disadvantage is that it is easily damaged in handling. It should therefore have a removable protective cover which can be installed for handling and removed prior to testing or flight.



MATERIAL: PLASTIC FOAM

TITLE: SLV-1B FOURTH STAGE PAYLOAD ADAPTER SECTION	DRAWN: ESC	APPROVED:	SHEET 1 OF 1	FILE NUMBER
	CHECKED: CVB	DATE: 10 Jan 61	SCALE: 1/8	D100-082-0

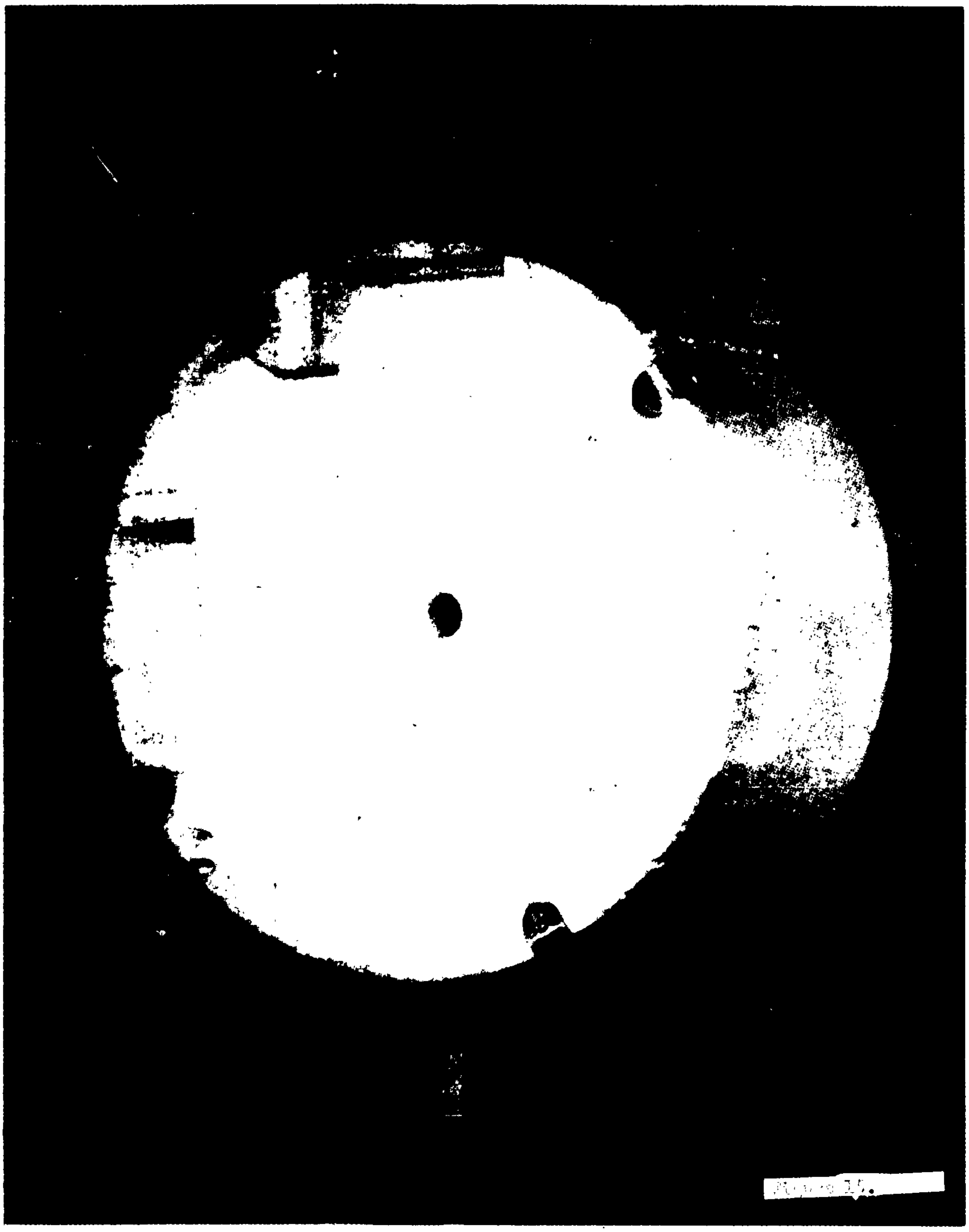
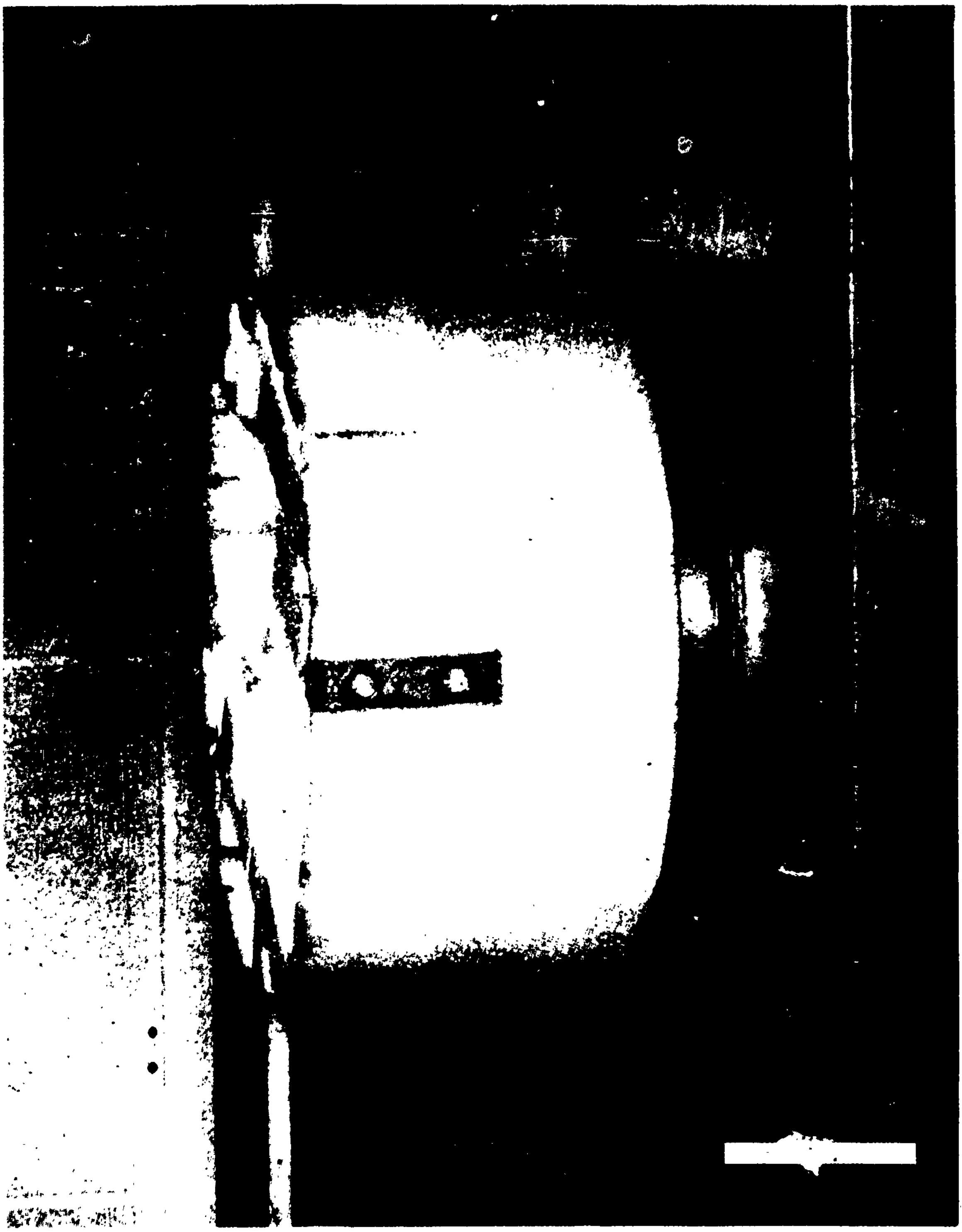


Figure 14.



3. The spacecraft structure should be removable from the payload adapter section. If the plastic foam is used, the following methods of mounting to the foam have been used successfully:

a. Magnesium brackets are inserted at regular intervals into the foam and are epoxied or otherwise affixed permanently to the foam. (See Figure 15.) The bearing area and number of brackets is chosen to conform to tensile and compressive foam strengths and expected loading under test conditions. A payload mounting plate may then be attached to the brackets by threaded bolts. If desired, the under surface of the payload mounting plate may be utilized for mounting telemetry components, batteries, etc., by milling the foam to fit. This arrangement also leaves the top side of the mounting plate free for mounting the experiment.

b. A circular right-angle flange, made to fit flush with the top and side surfaces of the foam is epoxied about the entire top surface periphery. This flange may then be tapped as desired for attachment of the payload structure. This procedure is slightly heavier than the bracket method (a above), but provides for greater strength. This method is recommended for payloads in the 50-pound class.

4. An environmental test jig, for attachment to vibration tables, centrifuges, etc., should be constructed to duplicate the flight-model payload adapter section. Figure 16 shows such a fixture constructed by adhering a plastic foam-type payload adapter section to a hemisphere. The hemisphere can then be attached to the test fixture as required.

E. Typical Payload Assembly Schedule.

On the following pages are listed significant steps in the payload design, fabrication, assembly, testing and fielding process. It will be noted that the schedule includes buildup and test of a prototype payload. Such a philosophy is not always followed, although a complete mockup is always considered desirable. The schedule is intended to provide a broad outline and basis for planning by experimenters. This schedule is based upon near-complete in-house design, fabrication, assembly and test. In the case of less complex payloads, extremely competent design, and prior experience in payload buildup, the schedule may be compressed into nine months.

TYPICAL 13-MONTH SLV-1B PAYLOAD DESIGN & ASSEMBLY SCHEDULE

MONTHS BEFORE LAUNCH	EVENT OR STATUS
13	Experiment Parameters & Requirements Established
12½	Instrument Inputs & Outputs Established
12	Telemetry System & Requirements Established
10½	Scientific Instrument Design Complete
10	Other Componenet and Subassembly Design Complete
7½	Instruments, Other Components & Subassemblies Built
7	Component & Subassembly Checkout Complete
7	Mechanical Design Complete
5½	Component & Subassembly Compatibility Checks Complete
4½	Prototype Payload Complete
4	Prototype Design Approval Environmental & Operational Tests Complete
3½	Design Freeze
2	Flight & Spare Payloads Built
1½	Flight Acceptance Environmental & Operational Tests of Flight and Spare Payloads Complete
1	Flight & Spare Payloads Ready to Go
½	Deliver Payloads to Launch Site
0	Launch

TYPICAL 13-MONTH SLV-1B PAYLOAD DESIGN & ASSEMBLY SCHEDULE

MONTHS BEFORE LAUNCH	STATUS
13	EXPERIMENT PARAMETERS & REQUIREMENTS ESTABLISHED. At the very beginning, the experimenter should define the scientific objectives in terms of apogee, geographic flight path, time of launch, hours of flight and other requirements which the shot must meet in order to be successful. Experimenter should immediately start the design of the scientific instruments to be flown, with first emphasis on the instrument/electronics interface requirements such as data rate, data shape, power input, number of channels required, etc. This should enable the electronics/logic/telemetry designers to begin their design at an early date.
12½	INSTRUMENT INPUTS & OUTPUTS ESTABLISHED. The experimenter should have available that information which will enable the design of the electronics, logic and telemetry to proceed, as stated above. In addition, he should have a good estimate on the physical size and weight of the instruments. Design of the electronics, logic, telemetry and mechanical configuration will begin as soon as the experimenter has furnished the required information.
12	TELEMETRY SYSTEM & REQUIREMENTS ESTABLISHED. At this stage, the type telemetry system will be decided on and its requirements will be established. Definition of the telemetry system will tie down power requirements and enable determination of battery weight. If long lead time telemetry items are to be used, procurement action should begin immediately. By this time, the physical aspects of the payload (i.e., weights, size of components, etc.) should be reasonably well established.

10½ SCIENTIFIC INSTRUMENT DESIGN COMPLETE. By this time, the scientific instruments should be completely defined insofar as shape, weight and other physical characteristics are concerned. Drawings should be complete and fabrication in progress or ready to begin. This should constitute a design freeze on these instruments, with the exception of minor revisions. Beyond this time, no changes should be made in the instruments which would affect their size, weight, power or data requirements or the electronics and telemetry. In addition, the experimenter should have all additional requirements for special items such as timers, change-overs, etc., definitely established.

10 OTHER COMPONENT & SUBASSEMBLY DESIGN COMPLETE. Electronics/logic/programmer circuitry should be in drawing form by now. "Breadboard" testing should be complete and the design should be at such a point that only minor, if any, revisions will be required from this point on. Fabrication and assembly of components and subassemblies will begin at the earliest practical date, and should already be in progress for some items.

7½ INSTRUMENTS, OTHER COMPONENTS & SUBASSEMBLIES BUILT. At this stage, all major items of the payload will be built.

7 COMPONENT & SUBASSEMBLY CHECKOUT COMPLETE. By now, all electrical and design approval testing of individual circuits, components and subassemblies should be complete. The project should be at such a stage that compatibility checks can get under way.

7 MECHANICAL DESIGN COMPLETE. A firm mechanical design should be in being now. Sizes, weights, locations and all other physical variables should be accurately known. A complete mockup of the entire payload should also exist. Wiring harnesses should be in progress using the mockup for sizing, routing, etc.

$5\frac{1}{2}$	COMPONENT & SUBASSEMBLY COMPATIBILITY CHECKS COMPLETE. At this time, sufficient checks and tests should have been made to insure compatible operation of subassemblies. As a final step in this phase of the operation, virtually the entire payload will have been assembled and operated. Except for minor changes, this should just about represent the final make-up of the payload.
$4\frac{1}{2}$	PROTOTYPE PAYLOAD COMPLETE. The prototype payload should be built and ready for operational and environmental checks.
4	PROTOTYPE ENVIRONMENTAL & OPERATIONAL CHECKS COMPLETE. The prototype payload shall have been subjected to the operational and environmental design approval test program by now.
$3\frac{1}{2}$	DESIGN FREEZE. After incorporating any changes deemed necessary as a result of the test program on the prototype, the payload design should be frozen.
2	FLIGHT & SPARE PAYLOADS BUILT. Flight and spare payloads will be built and ready for checkout at this time.
$1\frac{1}{2}$	FLIGHT ACCEPTANCE ENVIRONMENTAL & OPERATIONAL TESTS OF FLIGHT & SPARE PAYLOADS COMPLETE. By now, all flight acceptance tests, as specified by SWTTS will have been made. Payloads should be in a condition suitable for inspection by SWTTS at this time.
1	FLIGHT & SPARE PAYLOADS READY TO GO. Any last minute changes will have been completed by now and the payloads will be ready to go to the launch site.
$\frac{1}{2}$	DELIVER FLIGHT & SPARE PAYLOADS TO LAUNCH SITE. By now, all checkout gear, support equipment, etc., will have been prepared and shipped to the launch site.
0	LAUNCH. The last two weeks are spent in preflight checkout and integration of the payload and the missile.

A payload-to-vehicle fit-check is normally scheduled about four months prior to launch. It is therefore mandatory that finalized mechanical design and full scale mockup be in being at this time.

A monthly payload status report, to be submitted in two (2) copies to AFSWC (SWTTS) during the first week of each month, is required. A sample format for this report is shown on the following page. These reports are utilized to keep all cognizant agencies apprised of payload progress. These data, together with the vehicle delivery schedule, are utilized in establishing launch schedules.

F. Construction of Payload Control Console.

Control of the payload during checkout at the launch site, combined systems test, and launch will be the responsibility of the payload agency (with the exception of any ordnance items in the payload). The payload agency will therefore design, build, and deliver a payload control unit to perform the desired functions. All experimenters' external power supplies will be supplied by the experimenter and will be made a part of the console, unless a substitute item is available from the launch agency (advance requests for equipment of this type are covered in Section IV, Operations at the Launch Site). All switching functions will be powered from blockhouse power - the systems will be so wired as to preclude any switching without blockhouse power. Emergency OFF position on the Test Conductor's Console will turn all power to missile and payload off from either internal or external. A console is available in the blockhouse (AMR Complex 18A, CCMTA), which will accommodate a standard 19 x 17½ inch panel. The experimenters' control panel should be designed to be mounted in this unit. The blockhouse console is equipped with a MOPS (Operational Intercom) unit. Leads from the payload control console to the umbilical junction box should be 60 feet long and should be terminated in spade lugs which will accept 10-32 (.19 inch diameter) bolts. The payload control console should also be equipped with two function switches which can give remote light indications on the Test Conductor's Control Panel. These are:

PAYLOAD POWER	EXTERNAL-INTERNAL
PAYLOAD	READY-HOLD

PAYLOAD DESIGN AND ASSEMBLY SCHEDULE FOR ____

CALENDAR MONTH	SEP '62	OCT '62	NOV '62	DEC '62	JAN '63	FEB '63	MAR '63	APR '63	MAY '63	JUN '63	JUL '63	AUG '63	
EXPERIMENT PARAMETERS & REQUIREMENTS FIRM													
INSTRUMENT REQUIREMENTS ESTABLISHED													
TELEMETRY SYSTEM & REQUIREMENTS ESTABLISHED													
COMPONENT & SUBASSEMBLY DESIGN													
COMPONENT & SUBASSEMBLY CHECKS & TESTS													
MECHANICAL DESIGN COMPLETE													
COMPONENT & SUBASSEMBLY COMPATIBILITY TESTS INCLUDING ENVIRONMENTAL													
FINAL DESIGN FREEZE													
FLIGHT PAYLOAD BUILDUP													
FLIGHT PAYLOAD COMPLETE													
FLIGHT PAYLOAD ENVIRON & OPERNL TESTS													
PAYLOAD COMPLETE													

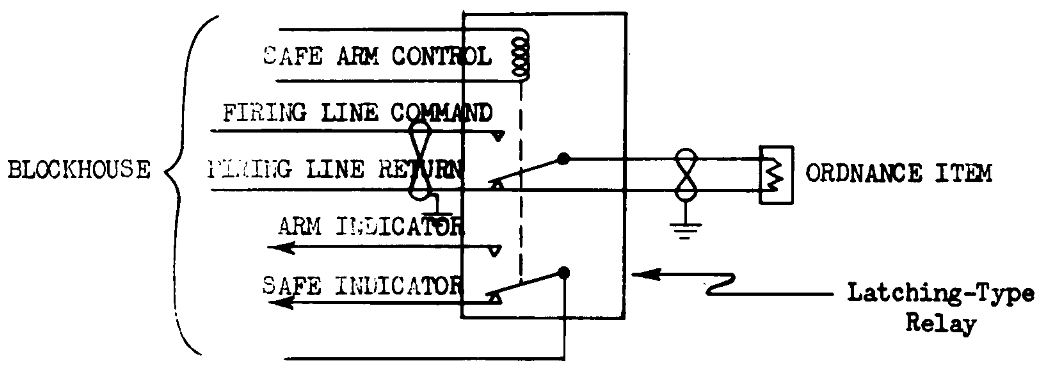
COMPLETED PORTION OF SCHEDULE

CURRENT AS OF ____________

Physical layout and electrical wiring diagrams of the payload control unit should be prepared for compatibility review by the launch agency at the earliest possible time.

It is anticipated that the same Payload Control Panel will be used to check out and control the experiment during its buildup and environmental test phase.

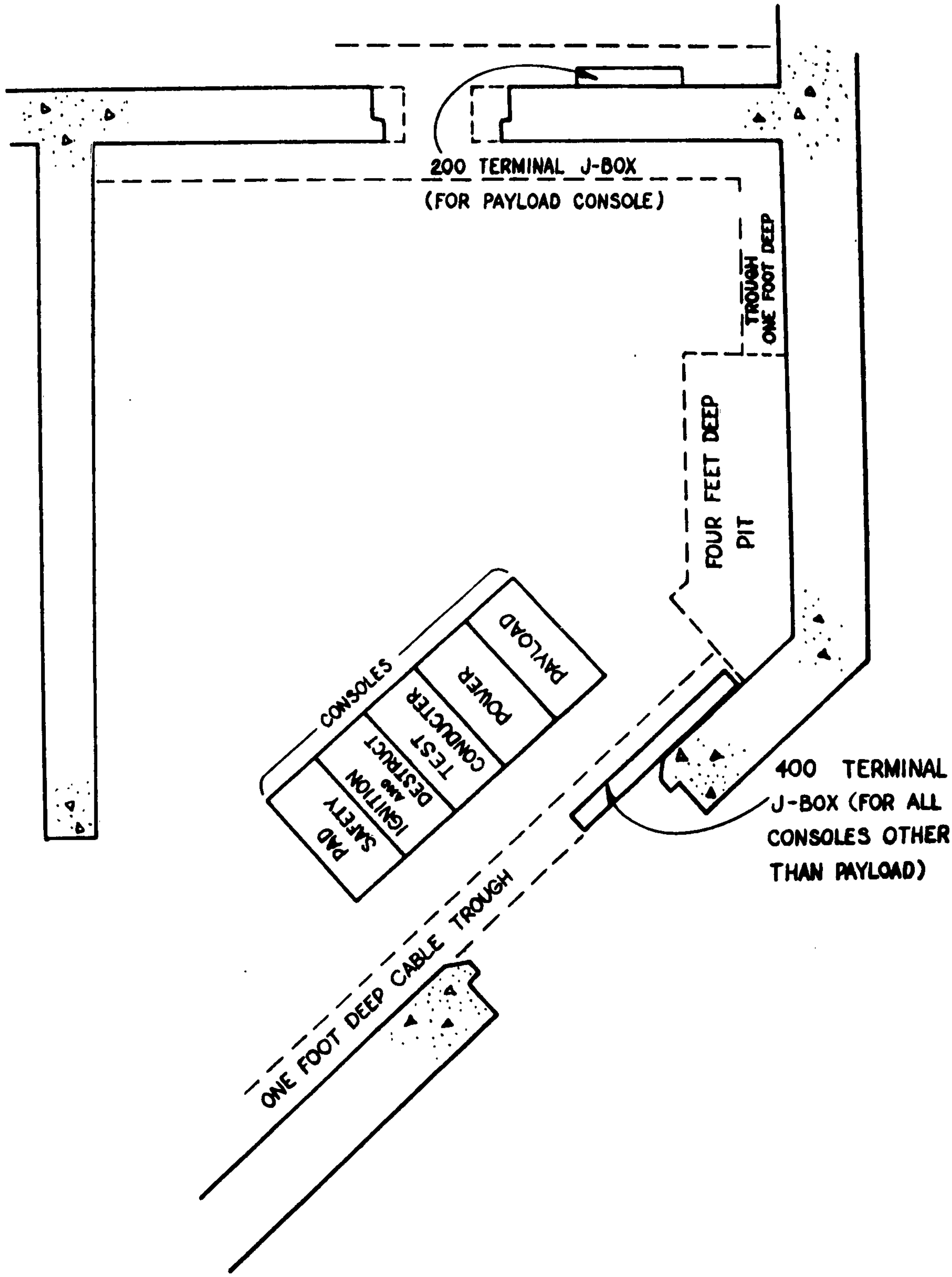
Ordnance items in the payload will be controlled directly by the Test Conductor or will be routed through the Test Conductor's panel to insure against accidental ordnance initiation. A payload arm/safe indicator and a payload arm-safe command switch will be located on the Test Conductor's console. Wiring to ordnance items should be shielded. A typical SAFE/ARM arrangement is shown below:



Location, type, purpose, procedures, and wiring of ordnance items in the payload should be clearly shown on early PR submissions. Payload ordnance items must meet "AFSC Interim Standards to Minimize the Hazards of Electromagnetic Radiation to Electroexplosive Devices" (1 watt-1 amp no fire characteristics or survival in 100 watt/m^2 field).

Console Array - Blockhouse 18A

SCALE: 1/4 IN. = 1 FT.



G. Applicable MIL Specifications, NASA Publications, and Other Documents.

On the following pages, a list of MIL Specifications, NASA publications, and other pertinent documents, is given. These specifications may be utilized in preparing procurement specifications, or applied to design and assembly. These specifications embody current standards in use today throughout the United States missile and space programs. Payloads fabricated for flight in the SLV-1B vehicle are expected to be constructed in accordance with accepted good practices and to utilize materials which conform to current standards compatible with the expected environment. Design Approval Testing and Flight Acceptance Testing procedures, outlined in Section II.E., are used to confirm the reliability of a payload.

MISCELLANEOUS DOCUMENTS APPLICABLE TO SPACE PROBE PAYLOADS

AFR 80-5, Reliability Program for Systems, Subsystems, and Equipments.

NASA Quality Publication 200-2, Quality Program Provisions for Space System Contractors.

NASA Quality Publication NPC-200-3, Inspection System Provisions for Suppliers of Space Materials, Parts, Components, and Services.

Inter-Range Instrumentation Group (IRIG) Publications available from Secretariat, IRIG, White Sands Missile Range, New Mexico or ASTIA:

IRIG Document No. 106-60, Telemetry Standards

IRIG Document No. 104-60, Standard Time Formats

IRIG Document No. 102-61, Standard Coordinate System and Data Format for Antenna Patterns

SUGGESTED MILITARY AND FEDERAL STANDARDS AND SPECIFICATIONS FOR SPACE PROBE PAYLOADS

MIL-A-9067C	Adhesive Bonding, Process and Inspection, Requirements For
MIL-A-14042A	Adhesive, Epoxy
MIL-A-52194	Adhesive, Epoxy, For Bonding Glass Reinforced Polyester
MIL-A-8623A(1)	Adhesive, Epoxy Resin, Metal-to-Metal Structural Bonding
FED-STD-175 TEST MD	Adhesives, Methods of Testing
MIL-B-16955(2)	Battery, Silver-Zinc, Secondary Type
MIL-C-20C SUPP 1	Capacitor, Fixed, Ceramic Dielectric, Temperature Compensating, General Specification For
MIL-C-11272B	Capacitors, Fixed, Glass Dielectric, General Specifications For
MIL-C-55057	Capacitors, Fixed, Solid Electrolyte, Tantalum
MIL-C-26655A(1)	Capacitor, Fixed, Solid Electrolyte, Tantalum, General Specifications For
MIL-C-26482	Connectors, Electric, Circular, Miniature, Quick Disconnect
MIL-C-5015D	Connectors, Electric, AN Type
MIL-C-25516	Connectors, Electric, Miniature, Shielded or Unshielded, Environmental Resisting Type, General Specifications For
MIL-C-21097A(1)	Connectors, Electric, Printed Wiring Board, General Purpose, General Specification For

MIL-C-22557	Connector, Electric, Miniature, For Radio Frequency Cables, General Specification For
MIL-C-3650A	Connectors, Coaxial, Radio Frequency
MIL-C-26500A	Connectors, Electric, General Purpose, Miniature, Circular, Environmental Resisting, 200°C Ambient Temperature
MIL-C-27239A	Connectors, Radio Frequency, General Requirements and Inspection
MIL-C-8384B	Connectors, Plug and Receptacle, Molded Body, and Accessories, General Specification For
MIL-C-12520B	Connectors, Plug and Receptacle, Electrical, Waterproof, and Accessories, General Specification For
MIL-E-25366A	Electric and Electronic Equipment and Systems, Guided Missiles, Installation of, General Specification For
MIL-E-7016C	Electric Load and Power Source Capacity, Analysis of, Method for Aircraft and Missiles
MIL-E-26144	Electric Power, Missile, Characterisitics and Utilization, General Specification For
MIL-E-1D	Electron Tubes and Crystal Rectifiers
MIL-STD-200E	Electron Tubes and Semiconductor Devices, Diode
MIL-E-22436	Electronic Assemblies, General Specification For
MIL-STD-439	Electronic Circuits
MIL-STD-242	Electronic Equipment Parts (Selected STDS)
MIL-E-8189B(1)	Electronic Equipment, Guided Missiles, General Specification For
MIL-E-5400D	Electronic Equipment, Aircraft, General Specification For

MIL-E-19600A	Electronic Modules, Aircraft, General Requirements For
MIL-STD-446(1)	Environmental Requirements for Electronic Component Parts
MIL-E-5272C	Environmental Testing, Aeronautical and Associated Equipment, General Specification For
MIL-E-5422D	Environmental Testing, Aircraft Electrical Equipment
MIL-I-8660A	Insulating and Sealing Compound, Electrical
MIL-I-14169	Insulating Compound, Electrical, Potting
MIL-I-631D	Insulation, Electrical, Synthetic Resin, Non-Rigid
MIL-I-7444B	Insulation Sleeving, Electrical, Flexible
MIL-I-23053	Insulation Sleeving, Electrical, Flexible, Heat-Shrinkable
MIL-I-22129B	Insulation Tubing, Electrical, Polytetrafluoroethylene Resin, Non-Rigid
MIL-I-26600	
MIL-M-45202	Magnesium Alloy, Anodic Treatment Of
QQ-M-38	Magnesium Alloy Die Castings
QQ-M-40A	Magnesium Alloy Forgings
QQ-M-44A(1)	Magnesium Alloy Plate and Sheet (AZ31)
MIL-M-3171A(1)	Magnesium Alloy, Process for Corrosion Protection Of
MIL-M-26075B	Magnesium Alloy, Sheet and Plate (HK-31A)
MIL-P-22693	Plastic Laminates, Glass Fabric Base, High Strength

MIL-P-21466	Plastic Sheet, Laminated, Copper Clad, Glass-Base Epoxy
MIL-P-55110	Printed Wiring Boards
MIL-STD-275A	Printed Wiring for Electronic Equipment
MIL-P-21193(1)	Printed Wiring, General Specification For
MIL-Q-9858	Quality Control Requirements
MIL-Q-21549A	Quality Control System Requirements
MIL-R-5757D SUPP 1	Relays, Electrical, Excluding Thermal, For Electronic and Communications-Type Equipment, General Specification For
MIL-R-6106C(1) SUPP 1	Relays, Electrical, Aircraft, General Specification For
MIL-R-25018(2)	Relays, Miniaturized, Hermetic Seal, Airborne Equipment, General Specification For
MIL-R-26667	Reliability and Longevity Requirements, Electronic Equipment, General Specification For
MIL-STD-441	Reliability of Military Electronic Equipment
MIL-R-25717C	Reliability Assurance Program for Electronic Equipment
MIL-R-22256	Reliability Requirements for Design of Electronic Equipment of Systems
MIL-R-26484	Reliability Requirements for Development of Electronic Subsystems or Equipment
MIL-R-11C(1) SUPP 1	Resistors, Fixed, Composition, Insulated, General Specification For
MIL-R-10509D(1)	Resistors, Fixed, Film, High Stability, General Specification For

MIL-R-93C	Resistors, Fixed, Wirewound, Accurate, General Specification For
MIL-R-9444A(3)	Resistors, Fixed, Wirewound, Precision High Temperature, General Specification For
MIL-R-18546C	Resistors, Fixed, Wirewound, Poner Type, Chassis Mounted, General Specification For
MIL-R-22097	Resistors, Variable, Rectilinear, Miniature
MIL-R-27208	Resistors, Variable, Trimmer

Semi-conductors, Diodes and Transistors are covered individually in the Series MIL-S-19500 and MIL-T-19500

MIL-S-12204B(1)	Solder, Aluminum Alloy
QQ-S-571C(2)	Solder, Lead Alloy, Tin-Lead Alloy, and Tin Alloy, Flux Cored Ribbon and Wire and Solid Form
QQ-S-561D(1)	Solder, Silver
MIL-S-45743	Soldering Electrical Connections, For Guided and Ballistic Missiles, with Electrically Heated Soldering Irons, and Resistance Soldering Apparatus, Procedures For
MIL-S6872A	Soldering Process, General Specification For
MIL-STD-440B	Soldering Techniques for Standard Type Solder Terminals
MIL-S-8169C	Specifications, Detail, Guided Missile, Preparation Of
MIL-STD-442A	Telemetry Standards for Missiles and Aircraft
MIL-T-18306A(1)	Test Equipment and Test Bench Harness Requirements for Avionic Equipment and Guided Missile Contracts
MIL-STD-202B Change 4	Test Methods for Electronic and Electrical Component Parts

MIL-T-18303	Test Procedures, Pre-Production and Inspection, For Aircraft Electronic Equipment, Format For
MIL-T-5422E(1)	Testing, Environmental, Aircraft Electronic Equipment

Transistors are listed individually in the MIL-S-19500 and MIL-T-19500 Series.

MIL-STD-701	Transistors

Wire and cable, electrical, are covered in a large number of MIL Specs, such as MIL-W-76, MIL-W-16878 Series, MIL-W-5086, MIL-C-7078, MIL-W-12349.

MIL-STD-252	Wired Equipment, Classification of Visual and Mechanical Defects
MIL-W-8160D	Wiring, Guided Missile, Installation Of, General Specification For

MARSHALL SPACE FLIGHT CENTER SPECIFICATIONS, STANDARDS AND PROCEDURES APPLICABLE TO SPACE PROBE PAYLOADS

Available From: Marshall Space Flight Center
National Aeronautics and Space Administration
M-MS-AD
Huntsville, Alabama

TITLE	NUMBER
ACCEPTABILITY LIMITS FOR ALL SELECTIVELY ETCHED ALUMINUM COMPONENTS	ABMA-STD-21
ACCEPTED STANDARDS OF ELECTRICAL ENGINEERING DESIGN	ABMA-STD-54
ALUMINUM ALLOY, PLATE AND SHEET (For Missile Use)	ABMA-PD-A-296
BRAZING, SILVER, SPECIFICATION FOR	10509307
CABLE AND HARNESS ASSEMBLIES, ELECTRICAL MISSILE SYSTEMS, GENERAL SPECIFICATION FOR	ABMA-PD-C-711
CAPACITORS, FIXED, CERAMIC-DIELECTRIC (General Purpose), GENERAL SPECIFICATION FOR	10M01185
CASTINGS, ALUMINUM AND MAGNESIUM ALLOY, RADIOGRAPHIC INSPECTION OF, ACCEPTANCE STANDARD FOR	MSFC-STD-100
COILS, RADIOFREQUENCY, AND TRANSFORMERS, INTERMEDIATE AND RADIOFREQUENCY, SPECIFICATION FOR	10M01191
CONNECTOR, PLUG AND RECEPTACLE PRINTED WIRING CIRCUIT BOARD, SPECIFICATION FOR	10M01188
CONNECTORS, ELECTRICAL, MS TYPE, SPECIFICATION FOR	10M01186
CONNECTORS, ELECTRIC, CIRCULAR, MINIATURE, QUICK DISCONNECT, SPECIFICATION FOR	10M01219
CONNECTORS, PLUG AND RECEPTACLE, ELECTRICAL (MOLDED BODY), AND ACCESSORIES, SPECIFICATION FOR	10M01187

CONNECTORS, RECEPTACLES, ELECTRICAL	MSFC-SPEC-119
DIRECT CURRENT AMPLIFIER MODULE, SPECIFICATION FOR	MSFC-SPEC-133
ENVIRONMENTAL PROTECTION OF VEHICLE AND SUPPORT EQUIPMENT COMPONENTS AND ASSEMBLIES, PROCEDURE FOR	10419900
INSULATION OF METALS IN MECHANICAL ASSEMBLIES, STANDARD FOR	10509316
LIGHTENING HOLES, ALUMINUM ALLOY, STANDARD FOR	10419958
MECHANICAL SYMBOLS	MSFC-STD-162
PACKAGING & PACKING OF PARTS, REPAIR PARTS, AND COMPONENTS FOR SPACE VEHICLES, GENERAL SPECIFICATION FOR	10509302
PLATING, CADMIUM (ELECTRO-DEPOSITED), SPECIFICATION FOR	10419960
PRINTED CIRCUIT DESIGN AND CONSTRUCTION STANDARD	ABMA-STD-428
RELAY ASSEMBLY, RELIABILITY TEST REQUIREMENTS	10M01032
RELAYS, HERMETICALLY SEALED, MISSILE EQUIPMENT	ABMA-PD-R-187
RESISTORS, FIXED, COMPOSITION (INSULATED), SPECIFICATION FOR	10M01168
RESISTORS, FIXED, WIREWOUND (ACCURATE), SPECIFICATION FOR	10M01196
RESISTORS, FIXED, WIREWOUND (POWER), SPECIFICATION FOR	10M01215
RESISTORS, VARIABLE, COMPOSITION, SPECIFICATION FOR	10M01172
RESISTORS, VARIABLE, RETILINEAR (ACCURATE), SPECIFICATION FOR	10M01195
RIVETING DESIGN, FABRICATION, AND INSPECTION, STANDARD FOR	10509301
RUBBER, SYNETHETIC AND RELATED ITEMS, AGE CONTROL OF, STANDARD FOR	10509311
SEALANTS, ORGANIC, APPLICATION OF, SPECIFICATION FOR	10509317

SEMI-CONDUCTOR DEVICES, SPECIFICATION FOR	10M01198
SOLDERING ELECTRICAL CONNECTIONS FOR SPACE VEHICLES, PROCEDURE FOR	10509300
SPHERES, FIBERGLASS, HIGH PRESSURE, SPECIFICATION FOR	10419907
WELDING, FUSION, SHIELDED ARC, MISSILE COMPONENTS, ALUMINUM AND MAGNESIUM, MANUAL OR AUTOMATIC	ABMA-PD-W-45

UNRELEASED (As of 1 Nov 61)

MARSHALL SPACE FLIGHT CENTER SPECIFICATIONS, STANDARDS AND PROCEDURES APPLICABLE TO SPACE PROBE PAYLOADS

Available When Released From: Marshall Space Flight Center
National Aeronautics & Space Administration
M-MS-AD
Huntsville, Alabama

TITLE	NUMBER
CABLE AND HARNESS ASSEMBLIES, ELECTRICAL (SPEC)	MSFC-SPEC-117
CAPACITOR, CERAMIC (RELIABILITY)	10M01228
CAPACITOR, FIXED, ELECTROLYTIC (TANTALUM) (RELIABILITY)	10M01225
CAPACITOR, FIXED, ELECTROLYTIC (TANTALUM) (SPECIFICATION)	10M01225
CLEANLINESS OF COMPONENTS (SPECIFICATION)	MSFC-SPEC-164
CONNECTORS, ELECTRICAL, PRINTED WIRING BOARD (SPECIFICATION)	10M01189
CONNECTORS, ELECTRICAL, M.S. TYPE (RELIABILITY)	10M01230
CONNECTORS, PLUG AND RECEPTACLE, ELECTRICAL (MOLDED BODY) & ACCESSORIES (SPECIFICATION)	10M01187
CONNECTORS, PLUG AND RECEPTACLE, ELECTRICAL (MOLDED BODY) AND ACCESSORIES (RELIABILITY)	10M01231
CONNECTORS, PLUG AND RECEPTACLE PRINTED WIRING BOARD (SPECIFICATION)	10M01188
CONNECTOR PLUG AND RECEPTACLE, PRINTED WIRING BOARD (RELIABILITY)	10M01232
CONNECTORS, RECEPTACLES, ELECTRICAL, REVISION A	MSFC-SPEC-119
CONNECTOR, PYGMY (RELIABILITY)	10M01263

HARNESS AND CABLE ASSEMBLIES, ELECTRICAL INSTALLATION (PROCEDURE)	MSFC-PROC-153
RELAYS (RELIABILITY)	10M01240
RESISTORS, FIXED, COMPOSITION (INSULATED) (RELIABILITY)	10M01242
RESISTORS, VARIABLE, RECTILINEAR (MINIATURE) (RELIABILITY)	10M01241
RIVETING, FABRICATION AND INSPECTION (STANDARD)	MSFC-STD-156
RUBBER AND RELATED ITEMS (STANDARD)	MSFC-STD-105
RUBBER, NATURAL AND SYNTHETIC (SPECIFICATION)	MSFC-SPEC-101
SEMI-CONDUCTOR DEVICES, GENERAL (RELIABILITY)	10M01250
SOLDERING, ELECTRICAL CONNECTORS (SPECIFICATION)	MSFC-SPEC-158
WIRING PROCEDURES, ELECTRIC (SPECIFICATION)	MSFC-SPEC-127

IV. FLIGHT PLANNING DOCUMENTATION.

A. The formal establishment of a program on a Range and the proper levying of requirements for Range support are among the most important "support" items to be accomplished for a launch. The discussion and schedules which follow apply, in particular, to the Atlantic Missile Range. The documentation system utilized is the National Range Documentation System (NRD), which applies to: Atlantic Missile Range (AMR), Pacific Missile Range (PMR), White Sands Missile Range (WSMR), and to the Service Ranges: Naval Ordnance Test Station (NOTS), Air Proving Ground Center (APGC), and the Air Force Flight Test Center (AFFTC). A more complete description of Atlantic Missile Range Support capabilities and the procedures by which these capabilities are provided is contained in "Range Users' Handbook, Atlantic Missile Range". Authorized agencies may obtain a copy by writing to: AFMTC DCS Plans & Programs, Patrick AFB, Florida.

AFSWC (SWTTS) has the responsibility for submitting all required documentation for the SLV-1B launches; i.e., all "Range User" documentation. (See the documentation flow lines on the chart shown on the following page.)

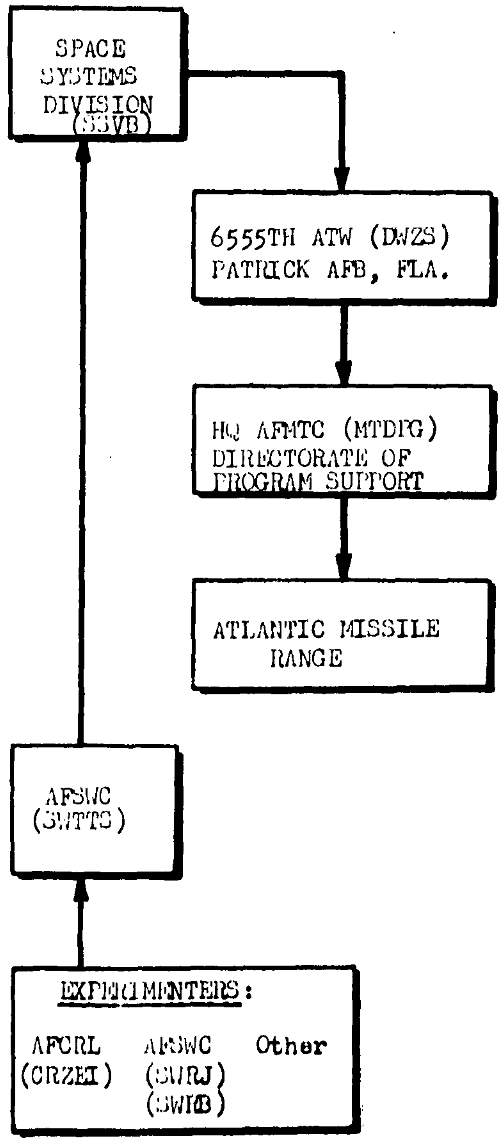
Only those documents which require experimenters inputs or directly affect the support of a given experiment are detailed herein.

B. The principal documents necessary for proper establishment and support of a program at the Range are:

1. Planning Estimate (PE). The PE presents a brief picture of the total program, stating general items for AFMTC long-range planning. This document will be prepared and submitted by AFSWC (SWTTS) with inputs as appropriate from the Program Management Office and experimenters. Known long lead-time items, such as special instrumentation or special launch facilities, should be mentioned in the PE.

2. Program Requirements (PR). The SLV-1B program utilizes a split-requirements document. The Booster Requirements Document (BRD) is prepared by AFSWC (SWTTS) for the standard SLV-1B booster. Each flight is then further documented with specific payload requirements by submission of a PR. AFSWC (SWTTS) will assist the experimenter in listing his requirements on the standard NRD PR forms. Initial listing of data on these forms is the responsibility of the experimenter.

SLV-1B SPACE PROBE
AMR RANGE DOCUMENTATION FLOW CHART



The PR is normally submitted 18 months prior to the first launch, but may be submitted as late as 6 months if no new facilities, training, funding, etc., are required. Therefore, the PR should be submitted as soon as information on the payload is available and revised later, if necessary. The PR is the most important (and lengthy) document in which the experimenter is directly involved. It is suggested that the experimenter arrange a meeting with AFSWC (SWTTS) at the earliest time for the purpose of receiving instruction and assistance in PR preparation to meet the AFMTC submission date. AFMTC Pamphlet 57-2, Program Requirements Handbook, gives instructions in detail for filling out PR pages. This handbook is available from AFMTC (MTDA), Patrick AFB, Florida.

3. The Program Support Plan (PSP) outlines the support that can be provided by the AFMTC support organizations to meet Range User requirements as delineated in the PR.

4. The Operations Requirement (OR) describes in complete detail the requirements necessary to accomplish a specific test or series of tests in the overall test program and is prepared by AFSWC (SWTTS). The accepted OR allows the Range User to pass on test requirements to AFMTC support organizations and to obtain support from AFMTC.

5. The Operations Directive (OD) is the official AFMTC publication which mobilizes the resources of AFMTC to support a launch.

C. Other Required Documents:

1. The System Test Objectives (STO) is prepared for SSD by AFSWC (SWTTS). Detailed information necessary for preparation of this document will be requested from experimenters about 5 months prior to launch. Basic data required will be the following:

a. Objectives of the experiment and basic requirements to accomplish these objectives (time above a certain altitude, basic measurements to be made, etc.).

b. GO-NO-GO Conditions. Consider all sensors and all data outputs. Which of these constitute a NO-GO situation if failure is detected during the pre-launch sequence? List specific limits for voltages, currents, etc.

c. Payload countdown functional checks and planned procedures during the countdown.

2. The Range Safety Report (RSR) is prepared by AFSWC (SWTTS) and contains a detailed analysis of such considerations as trajectories, dispersion patterns, destruct dynamics, and failure turning rates during powered and coasting flight. Special circumstances may require an analysis of reentry heating of the payload, in which case experimenters will assist by supplying information on the material composition, physical arrangement, and dimensions of the payload. This information, if needed, will be requested approximately four months prior to launch. This report is due at AFMTC at least 30 days prior to launch.

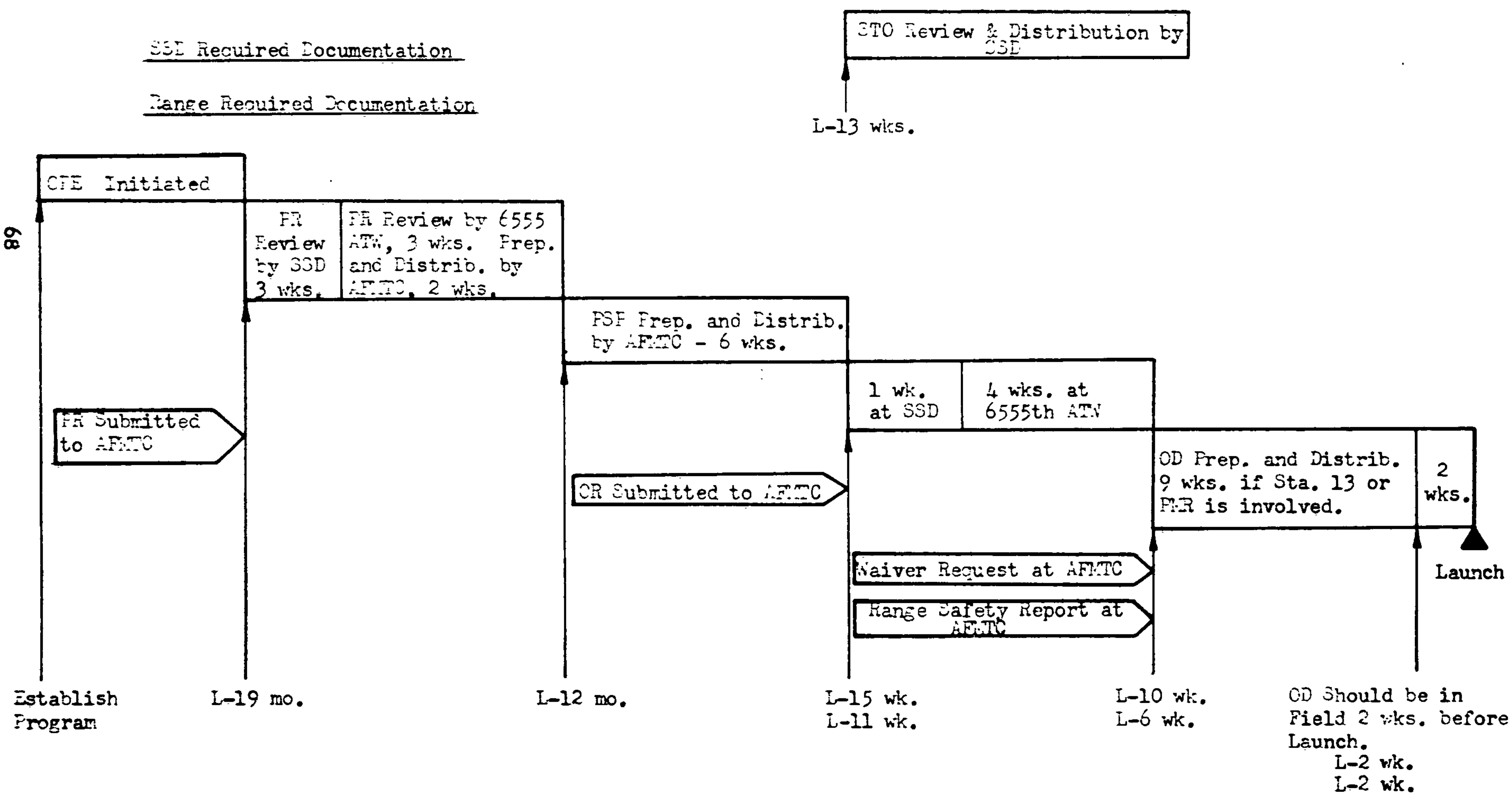
3. Several other documents and reports are prepared before and after a SLV-1B launch. In general, the experimenter will be contacted if data are required for any of these reports.

D. It should be noted that a delay in the submission of any of the above documents could result in the launch date being slipped (or the assignment of another payload to the vehicle).

E. Security Classification. The security classification for the SLV-1B vehicle is defined in the SSD Document entitled "Security Classification Guide - Blue Scout Program", dated 23 Feb 1962, as revised by letter changes dated 19 June and 12 Dec 1962. Launch schedules, payload technical information and payload objectives are specified by the payload agency or program office and are classified individually depending upon the experiment.

LEAD TIME FOR SLV-3 DOCUMENTATION

NOTE: First dates are for maximum lead time.
Second dates are for minimum lead time.
Dates are submission dates to AFMTC (via SSD).



V. OPERATIONS AT THE LAUNCH SITE.

Prior to delivery of any payload to the launch site, all environmental and functional checks must be satisfactorily completed and the payload certified to be environmentally, functionally, and dimensionally compatible with the vehicle. All sensor calibrations should have been completed prior to shipment. The payload is essentially in a flight-ready condition with the possible exception of flight-battery processing and installation.

A. Approximate Schedule.

Upon arrival of the payload and accompanying personnel, the following schedule can normally be anticipated:

Working Days	Function
F-20	Payload arrives in Hangar AA or the Combined Systems Test Building.
	Payload is received and inspected.
	Payload is set up for checkout.
	Battery processing is started.
	Payload is checked out.
	Payload adapter is bonded to the fourth-stage motor in the Missile Ordnance Assembly Building.
F-15	Payload is moved to the Combined Systems Test Building for subsystems tests.
F-12	Payload is moved to Dynamic Balancing Facility.
	Payload and fourth-stage motor are mated.
	Third and fourth-stage motors are mated.
	Dynamic balancing is performed.
F-7	The third and fourth stages are moved to the Missile Ordnance Assembly Building for final vehicle assembly.

Complete vehicle is assembled on transporter.

F-6 Assembled vehicle is moved on transporter to the Combined Systems Test Building (CSTB).

Systems test is performed on vehicle in CSTB.

F-3 Vehicle is moved to pad on transporter and compatibility checks are accomplished.

F-1 Dry run of launch countdown (F-1 day check) is accomplished. Ordnance is installed.

F-0 Vehicle is launched.

The actual launch date will be chosen at the Range Scheduling Meeting on Thursday, preceding the week of the launch.

B. Vehicle Assembly Flow Diagram.

The normal vehicle assembly will proceed according to the charts shown on the following pages. Primary items in the schedule (which are combined operations) affecting the payload agency are:

1. Delivery of the payload adapter section.

2. Spin balance.

3. Combined systems tests.

4. Pad compatibility checks.

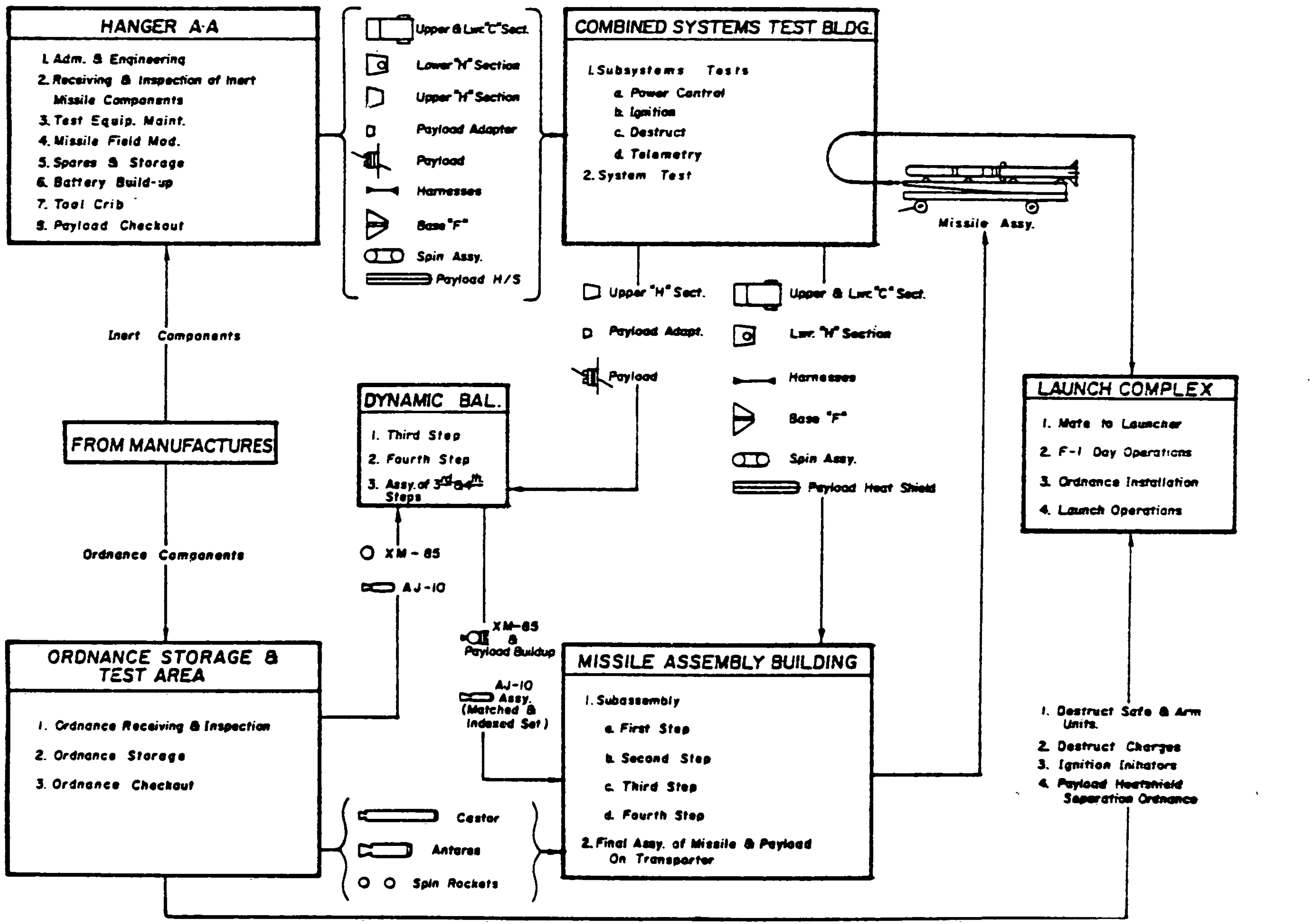
5. F-1 day dry run.

6. Launch.

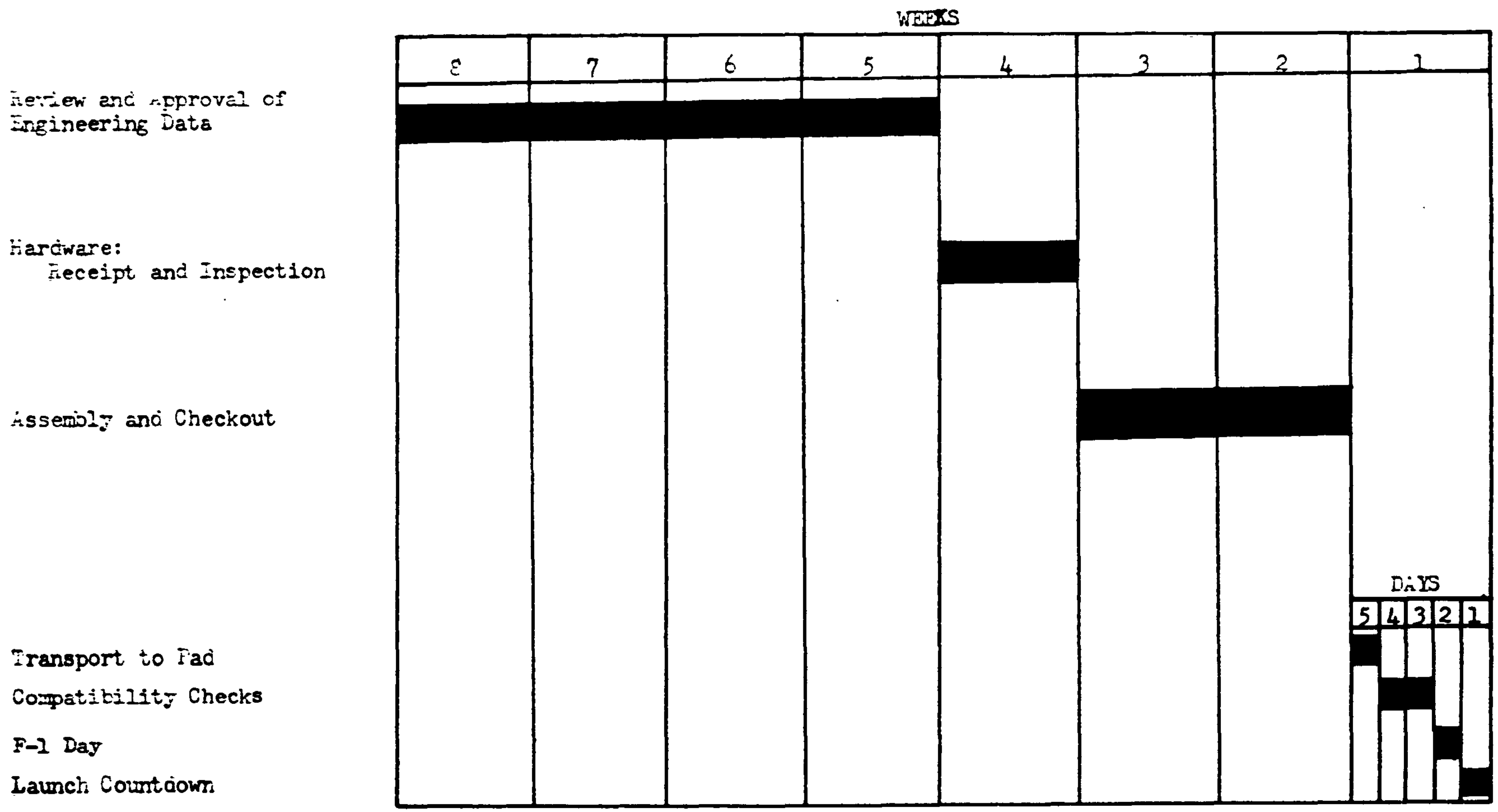
C. Available Checkout Equipment and Facilities:

The diagram on the following page illustrates telemetry check-out equipment available in the Combined Systems Test Building. The station consists essentially of minimum equipment necessary to perform checkouts on a standard FM/FM system. The checkout station is equipped with a tape transport, oscilloscope, recording oscillograph (0-1000 cps), strip chart recorder (0-100 cps) and other miscellaneous test equipment to maintain the station. Any special equipment required by the experimenter (such as special receivers) must be supplied by the experimenter.

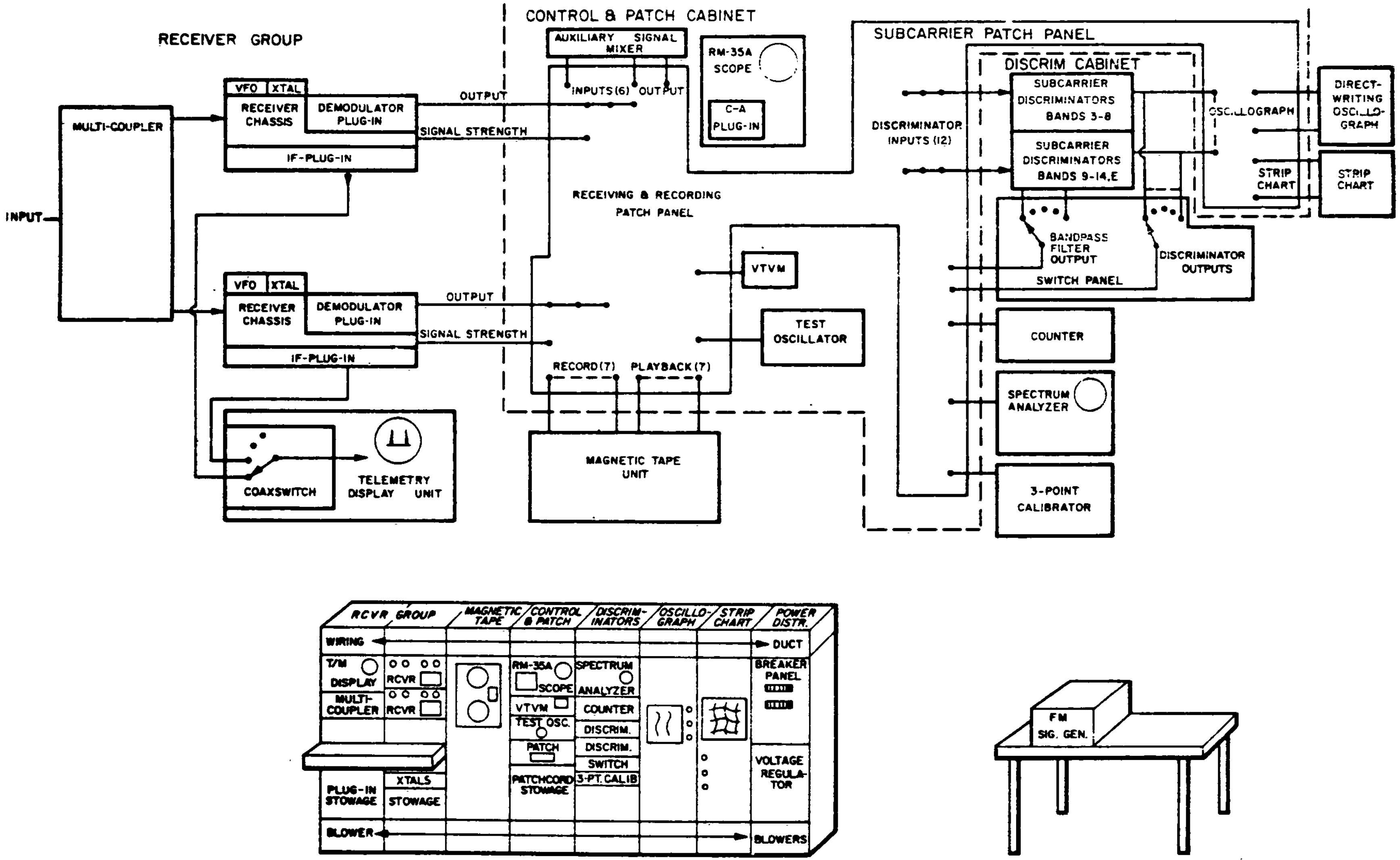
SLV - 1B FLOW DIAGRAM



AMR MISSILE PROCESSING SCHEDULE



SLV - 1B TELEMETRY GROUND CHECKOUT STATION



A battery charging facility, tool crib, machine shop, and limited test equipment are available in Hangar AA. Adequate bench space and power requirements will also be assigned by the launch agency (6555th Aerospace Test Wing, DWZS, Patrick AFB). It is suggested that the experimenter come to the launch site prepared to operate independently, except for a telemetry checkout station and those common items such as bench space and power. Necessary test equipment, meters, hand tools, spare parts, etc., should be brought to the launch site with the experimenter. Specific items programmed in the PR will be available.

D. The Flight-Test Working Group.

For each launch, a Flight-Test Working Group (FTWG) will be formed. The composition, duties, and responsibilities of this group are given in the SLV-1B Flight-Test Working Group Charter on the following pages.

E. Duties and Personnel Location During the Launch.

During the Combined Systems Tests, Pad Compatibility Checks, F-1 Day Dry Run, and Launch, the payload agency will be expected to perform the following functions:

1. Operate the Payload Control Console.

2. Provide a person or persons at Telemetry Receiving Station for assessment of received data and determination that payload is operating properly, both pre-launch and post-launch.

3. Provide a technical consultant to assist the Test Conductor (especially during the launch sequence) in deciding whether or not to launch under certain conditions (an inoperative down-range telemetry station for example).

Personnel in the blockhouse will be held to an absolute minimum. Personnel at the Range Telemetry Receiving Station will be held to that absolutely necessary to assess data and advise the blockhouse payload representative of the payload status.

Personnel not involved directly in the operation will stand by at Hangar AA, or other suitable location. Standby at Hangar AA is advised because of the communications available there with the launch operation.

SLV-1B
FLIGHT TEST WORKING GROUP CHARTER

14 Nov 62

1. INTRODUCTION.

A group of Air Force and Contractor technical personnel will be formed at AFMTC for the purpose of providing rapid appraisal and solution of technical problems in the field.

2. MISSION.

The Flight Test Working Group will be the coordinating agency for the test operation and technical aspects of test planning, launch, and evaluation at AFMTC.

3. SCOPE.

Following are typical test planning and evaluation functions assigned to the Flight Test Working Group:

a. Test Planning

Review documentation, i.e., the System Test Plan, Booster Requirements Document, and Operations Requirement.

Review and approve Test Countdown documents.

Review field design changes in the configuration of missile and support equipment, with particular emphasis on interface problems, instrumentation, end item configuration, test procedures and data requirements, and make recommendations, when applicable to the success of the mission, to the Chief of the SLV-1B Division.

Review master schedules, issued by the Program Office, and propose intermediate schedule dates. Review deviations from the master schedule necessitated by Range schedule interferences, weather and other unpredictable causes, and propose revised dates.

Provide technical support to the Project Division in the preparation of the Operations Requirement (OR) and in other matters concerning relationships with the Atlantic Missile Range.

b. Test Evaluation.

Accomplish post-test evaluation, using quick-look data.

Review significant pre-flight test results, and provide recommendations on missile readiness for subsequent testing and flight.

Participate in a post-flight critique and provide inputs for the Flash Report.

Assist in preparation of the Flight Test Report, review the report for concurrence and make recommendations for future tests.

4. ORGANIZATION.

The following agencies and/or offices will each provide a member to the Flight Test Working Group:

Air Force Special Weapons Center (SWTTS)

OAR Field Office (MTQU) (to include OAR Experimenter Agency)

Materiel Office (DWM)

Program Management (MTVPS)

Air Force Systems Command Test Site Office (REEPA)

Chance Vought, Astronautics Division

Experimenter Agency (other than OAR)

AFMTC Program Support (MTDPG)

Additional representation will be provided by the associated agencies as required by the Agenda; these representatives will work through the Flight Test Working Group member.

The Chief, SLV-1B Division, will be the Chairman of the Flight Test Working Group.

A representative of AFMTC and DWL will be invited to attend meetings as required.

Ad Hoc sub-groups will be established, as required, for investigation and recommendations on special problems.

The established organization structure has been reviewed and concurred with by AFSWC, OAR-FO, AFMTC, and Chance Vought/Astronautics.

5. RESPONSIBILITY.

The Flight Test Working Group will serve in an advisory capacity to the Chief, SLV-1B Division.

6. MEETINGS.

Meetings will be called by the Chairman, as required.

VI. APPLICABLE AGENCIES IN THE SLV-1B PROGRAM.

The SLV-1B vehicle is unique in the USAF inventory in that it is a complete "Blue Suit" operation from procurement through publication of the final flight report. Applicable agencies and their responsibilities in the SLV-1B Program are:

1. Space Systems Division, AFSC (SSVB) — OS 9-4661, Ext ~~3436~~ 3365
   SLV-1 Directorate — Area Code 213
   AF Unit Post Office
   Los Angeles 45, California

AFSSD has been designated as overall Program Administration Manager for the SLV-1B, as pertains to systems engineering, technical direction, and all other services required to provide SLV-1B space booster vehicles and to place designated experiments in the specified trajectories. A large portion of this effort is procurement of space booster vehicles, spares, Aerospace Ground Equipment (AGE), and related services.

2. Air Force Special Weapons Center (SWTTS) — CH 7-1711, Ext 3034
   Space Vehicle Br, Test Div, Test Dir — Area Code 505
   Kirtland Air Force Base
   Albuquerque, New Mexico

AFSWC (SWTTS) accomplishes payload liaison, flight planning documentation, and general technical and engineering assistance to experimenters, AFSSD, and the launch agency.

3. 6555th Aerospace Test Wing (DWZS) — UL 7-6050
   SLV-1B Project Office — Cocoa Beach Area Code 305
   Patrick AFB, Florida

The 6555th ATW provides a "Blue Suit" vehicle assembly, launch, and test evaluation capability at the Atlantic Missile Range.

4. 6595th Aerospace Test Wing (VWZB) — RE 4-4311, Ext 85865
   SLV-1 Project Office — Santa Maria Area Code 805
   Vandenberg AFB, California

The 6595th ATW provides a "Blue Suit" vehicle assembly, launch, and test evaluation capability at the Pacific Missile Range.

A Memorandum of Agreement between participating agencies in the SLV-1B AMR Launch Program is shown on the following pages.

MEMORANDUM OF AGREEMENT AMONG AFSSD, 6555TH ATW AND AFSWC
PERTAINING TO THE SUPPORT OF SLV-1B VEHICLES LAUNCHED
FROM THE ATLANTIC MISSILE RANGE

1. REFERENCE: Letter, SSV, AFSSD, subject: "AFSWC Participation in Support of SLV-1B Vehicles", dated 9 Jan 1962, and Staff Study, DWZ, 6555th ATW, subject: "AMR SLV-1B Follow-On Program", dated 16 Jul 1962.

2. PURPOSE: To define and record agreements between the Air Force Space Systems Division (AFSSD), the 6555th Aerospace Test Wing (ATW), and the Air Force Special Weapons Center (AFSWC), pertaining to the support of SLV-1B vehicles launched from the Atlantic Missile Range (AMR). Specifically, this agreement pertains to the support of SLV-1B vehicles, Numbers AD 622, AD 623, AD 624, AD 625, AD 626, AD 628, and AD 6214.

3. DEFINITIONS: For the purpose of this memorandum, the terms listed below are defined as follows:

a. Payload. Consists of the scientific sensors, associated supporting structure, telemetry set, power supplies and such electro-mechanical components as may be needed for control of the payload package.

b. Flight Planning. Consists of those documents required to establish and provide for a missile launch program at the Atlantic Missile Range. These documents include the following:

(1) Operational Program Estimate (OPE).

(2) Program Requirements Documents - to include Booster Requirement Document (BRD) and Program Requirements (PR).

(3) Operations Requirements Document (OR).

(4) Systems Test Objectives (STO).

(5) Flight Termination Systems Proposals.

(6) Flight Termination System Waiver Requests.

(7) Aerodynamic and Trajectory Documents.

(8) Quick-Look and Post-Flight Data Return Requirements.

c. Payload Integration. Consists of compatible installation of a complete payload package into the vehicle. The following tasks are included in payload integration:

(1) Standardized Specifications for Payload Tests.

(2) Establishing a System for Reporting Payload Environmental Tests.

(3) Determination of Payload/Vehicle Interface.

(4) Determination of Interface Requirements (Thermal, Electrical and Mechanical) and Establishment of Design Specifications.

(5) Incorporation of Payload Requirements into Aerospace Ground Equipment and Pre-Launch Operations and Schedules.

(6) Establish a Quality Assurance Program Commensurate with the Accepted Manufacturing Practices in the Military and Missile Industry.

(7) Assuring that Proposed Telemetry Systems are Compatible with Existing or Programmed Atlantic Missile Range Support Instrumentation.

d. Technical Field Representation and Support. Pertains solely to that associated with the booster vehicle. This consists of resident or on call consulting service in any one or all of the following categories:

(1) Propulsion.

(2) Command Destruct System.

(3) Aerospace Ground Equipment.

(4) Booster Vehicle Hardware.

(5) Booster Vehicle Instrumentation.

(6) Ignition System.

e. Coordination Services. Services necessary to assure the experimenter and the launch team have the appropriate vehicle technical and design data and vehicle assembly and launch procedures. These include:

(1) Coordination and Consolidation of Atlantic Missile Range Support Requirements.

(2) Liaison between Vehicle Contractor and Experimenter.

(3) Coordinate Aerospace Ground Equipment and Facilities Requirements for Payload Check-out.

(4) Provide the 6555th ATW with Contractor Vehicle Drawings, Technical Data, Assembly and Launch Procedures.

f. "Blue Suit" Launch Capability is defined as the capability of Air Force engineers (officers and civil service) and airmen technicians to receive, assemble, check-out and launch the SLV-1B vehicle.

g. "Blue Suit" Evaluation Capability is defined as the capability of Air Force (officers and civilians) to analyze and evaluate vehicle sub-systems and complete vehicle systems performance using data obtained during component-combined systems and flight tests. This evaluation will result in a formal report in sufficient copies to provide all interested agencies with information on each flight.

h. Technical Assistance. The service provided to the program manager pertaining to the engineering and technical aspects of the program. This assistance may be provided by direct contact with the contractor or payload agency; however, the program manager will be informed of all actions involving the contractor. This assistance will include such things as:

(1) Recommendations on vehicle and sub-systems design.

(2) Maintaining a file of contractor-furnished drawings and technical data and providing this information to agencies designated by the program manager.

4. MANAGEMENT & SUPPORT.

a. General. AFSSD is responsible for program management as pertains to systems engineering, technical direction and all other services required to provide SLV-1B space booster vehicles and to place designated experiments in the specified trajectories. The 6555th ATW is responsible for providing local management of SLV-1B launch operations at AMR, and for liaison between the AFMTC and AFSWC/SSD. The AFSWC is responsible for the technical assistance of SLV-1B space booster vehicles. Engineering assistance and recommendations provided by the AFSWC will not be binding on the contractor until approved by AFSSD.

b. Specific.

(1) Included in the responsibilities of the following agencies are the tasks listed below:

(a) AFSSD.

1. Procurement, production and provision of space booster vehicles, spares, AGE, and assembly, check-out and launch procedures.

2. Necessary coordination with AFSWC and the 6555th ATW as pertains to flight planning, payload integration, technical management, technical field representation and support of SLV-1B launches.

3. Provisions for contractor engineering services. The systems contractor will have responsibility for vehicle and AGE systems design changes and procedure and drawing changes.

(b) 6555th ATW.

1. "Blue Suit" launch capability.

2. "Blue Suit" evaluation capability.

3. Providing payload check-out space and standard test equipment, when available, to the experimenter.

4. Installation and check-out, of contractor fabricated AGE equipment under the direction and quality control of the AGE contractor.

5. Coordinate recommended vehicle and AGE design changes with the AFSWC field representative and AFSSD.

6. Provide the AFSWC with the necessary AFMTC documents and forms that pertain to flight planning documentation and keep AFSWC informed on changes to AFMTC requirements, policies, and procedures.

7. Assist AFSWC in preparation of the Flight Test Engineering and Analysis Report.

8. Preparation of the Flash Report, Follow-On Report and Final Launch Report.

(c) AFSWC.

1. Flight Planning, transmitted to SSD for submission to the Range.

2. Payload Integration.

3. Coordination Services.

4. Engineering assistance in the standardization of vehicles ar i spares.

5. General engineering assistance to AFSSD for contract management and launch support.

6. Provide minimum vehicle instrumentation as defined in previous correspondence with AFSSD.

7. Technical field representation and support.

8. Assist 6555th ATW in evaluation of quick-look data and preparation of Flash Report, Follow-On Report and Final Launch Report.

9. Preparation of Flight Test Engineering and Analysis Report.

5. Cognizant offices of each agency participating in this memorandum of agreement are as follows:

a. AFSSD (LtCol Donald A. Stine, Director)
Standard Launch Vehicle I (SSVB)

b. 6555th Aerospace Test Wing (LtCol Anthony A. Gomes, Chief)
SLV-1B Division

c. Space Vehicle Branch (Maj Gwynne W. White, Chief)
Test Division, Test Directorate
Air Force Special Weapons Center

APPROVED 26 Sep 62	APPROVED 10 Oct 62	APPROVED 31 Oct 62
THOMAS W. MORGAN LtColonel, USAF Deputy, Space Systems 6555th Aerospace Test Wing	TYLER A. REDFIELD Colonel, USAF Chief, Operations Division DCS/Plans & Operations	DONALD A STINE LtColonel, USAF Dir, Stan Launch Veh I Deputy for Engineering

APPENDIX A

TYPICAL YARDNEY PM SILVERCEL DATA

	PM-05	PM-1	PM-2	PM-3	PM-5	PM-10	PM-15	PM-16	PM-40	PM-58	PM-200
Nominal capacity (amp-hrs)	0.5	1.0	2.0	3.0	7.5	12.0	15.0	20.0	40.0	70.0	200.0
TYPICAL APPLICATION DATA											
60-min rate discharge											
Discharge rate (amps)	0.5	1.0	2.0	3.0	5.0	10.0	15.0	16.0	40.0	60.0	65.0**
Ampere-hr output*	0.85	1.95	3.5	4.6	9.5	15.5	18.5	26.5	47.0	87.0	293.0
Average voltage	1.46	1.47	1.48	1.49	1.49	1.48	1.47	1.49	1.48	1.48	1.50
Watt-hrs per pound*	25	42	33	34	49	44	44	49	45	61	80
Watt-hrs per cubic inch*+	1.5	2.3	2.0	2.3	4.8	2.8	3.0	3.5	2.9	4.3	5.6
PHYSICAL CHARACTERISTICS											
Maximum weight, filled (oz)	0.8	1.1	2.5	3.2	4.6	8.5	10.0	13.0	25.0	34.0	88.0
Overall height (inches)	1.56	2.02	2.55	2.89	2.91	4.81	4.94	6.13	7.09	7.25	10.44
Overall volume incl. terminals (cubic inch)	0.91	1.18	2.58	2.94	4.78	8.26	9.13	11.4	22.8	29.9	78.0
Volume less terminals (cu. in.)	0.70	0.98	2.19	2.5	4.0	7.35	7.8	10.0	20.6	27.6	70.7
Height less terminals (inches)	1.2	1.69	2.16	2.49	2.49	4.28	4.22	5.38	6.38	6.69	9.50
Width (inches)	1.08	1.08	1.72	1.72	2.08	2.32	2.31	2.3	3.25	3.25	4.19
Depth (inches)	0.54	0.54	0.59	0.59	0.79	0.74	0.80	0.81	0.99	1.27	1.78

**This model designed for 2-hr discharge rate or lower (3-hr rate shown here).
*To a final voltage of 1.1 V.
+Calculated using overall volume.

APPENDIX B

APPENDIX B

The Digilock Telemetering System is essentially a PCM/PSK system utilizing bi-orthogonal word coding to achieve a high communications efficiency. Data received at a receiver pre-detection signal-to-noise ratio of -6 db can be reconstructed to give a word error rate on order of 10^{-6}.

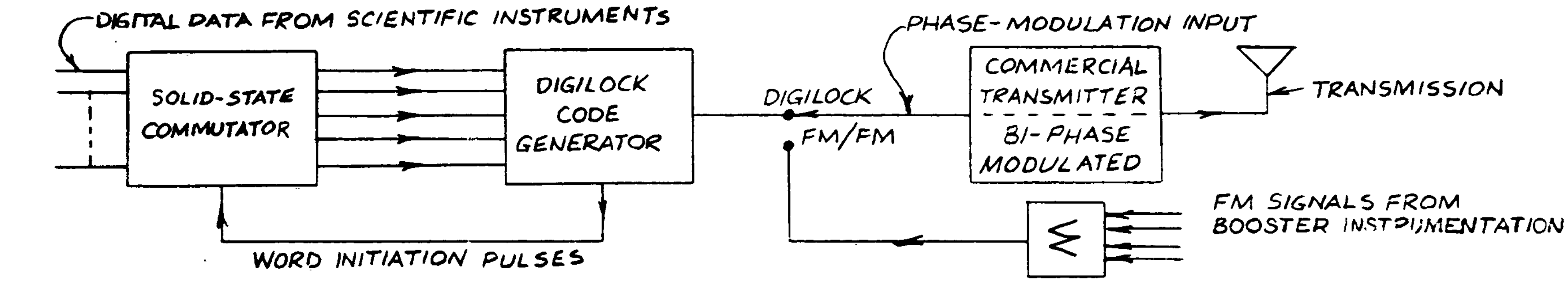
The Flight Encoder unit, developed by Space General Corporation, 9200 East Flair Drive, El Monte, California, for AFSWC under Contract AF 29(601)-2801, accepts 100 digital inputs. These inputs are read in words of 5 bits each by an internal electronic commutator, making a data frame of twenty 5-bit words. Each 5-bit word is encoded into a 16-bit "Digilock" word which has a mathematically orthogonal relationship to the other 31 possible words in the set. The Digilock words are transmitted serially, bi-phase modulating the transmitter. A Digilock "1" is transmitted as 10, a "0" is transmitted as 01. This insures that all bits have a zero amplitude average and that the phase-locked receiver tracking loop will always track the carrier, since the transmitted data possess no residual phase error over a sample of bits which is long compared to the time of one bit. System information rates of 64 bits per second and 512 bits per second are in use by AFSWC. These become transmitted bit rates of 428 and 3428 bits per second. Eight Digilock phase lock receivers, compatible with these bit rates, have been supplied to the Atlantic Missile Range by AFSWC on a permanent basis for support of any Digilock-equipped flight.

The Digilock system has been shown to have a very low error rate for $\beta = 8$.* Using $\beta = 80$ (10 db safety factor) and $H = 64$, it is found that a 250 milliwatt transmitter is quite suitable for transmission of data (at the 10^{-6} word error rate) from 50,000 n.m.

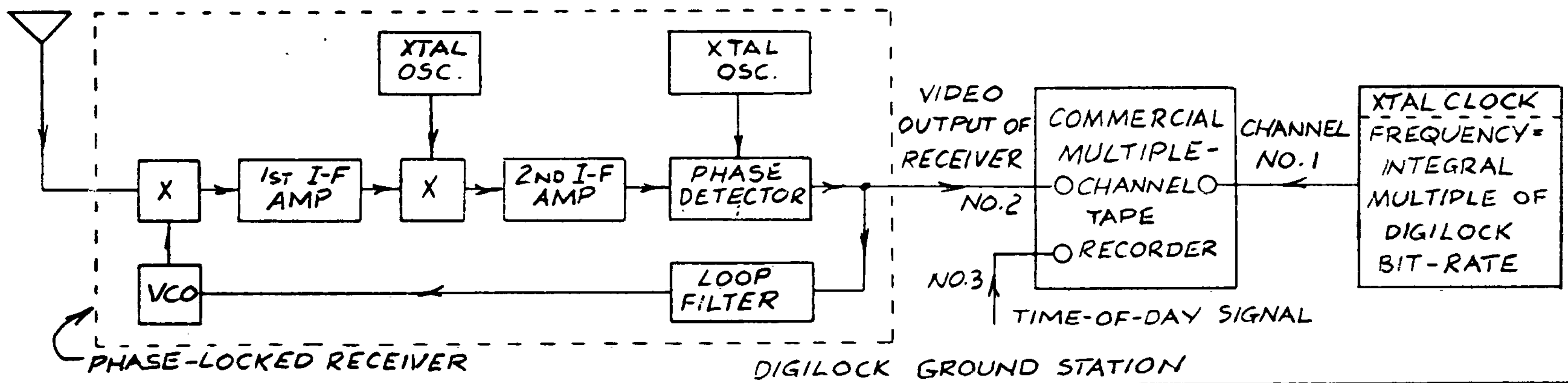
The data reduction process, at low S/N ratios (when the data no longer appear clearly digital), requires the use of an integrate-and-dump analog-to-digital converter driven by an external triggering signal. The data are converted to digital form and are processed into final format by a digital computer. These facilities are available at AFSWC.

Complete information on this system is available from AFSWC (SWTTS).

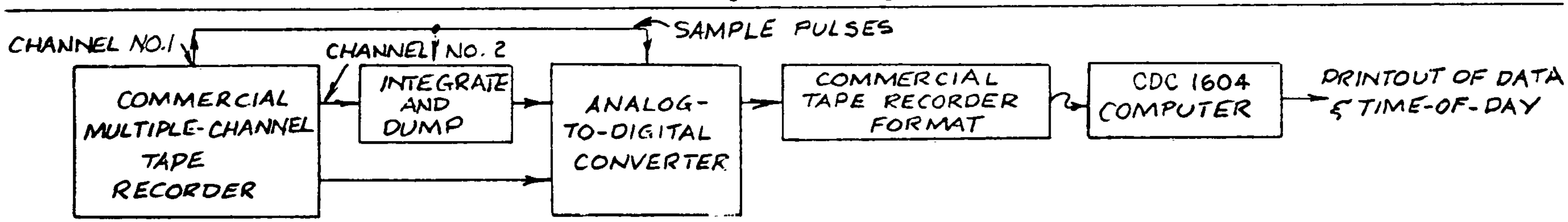
*1. Sanders, R. W., "Communications Efficiency Comparison of Several Communications Systems", PROC IRE: Vol 48, page 575-588, Apr 1960.
2. Jaffe, R. M., "Digilock Telemetry System for the AFSWC Blue Scout Junior", IRE TRANS on Space Electronics and Telemetry", Vol SET-8, page 44; Mar 1962.



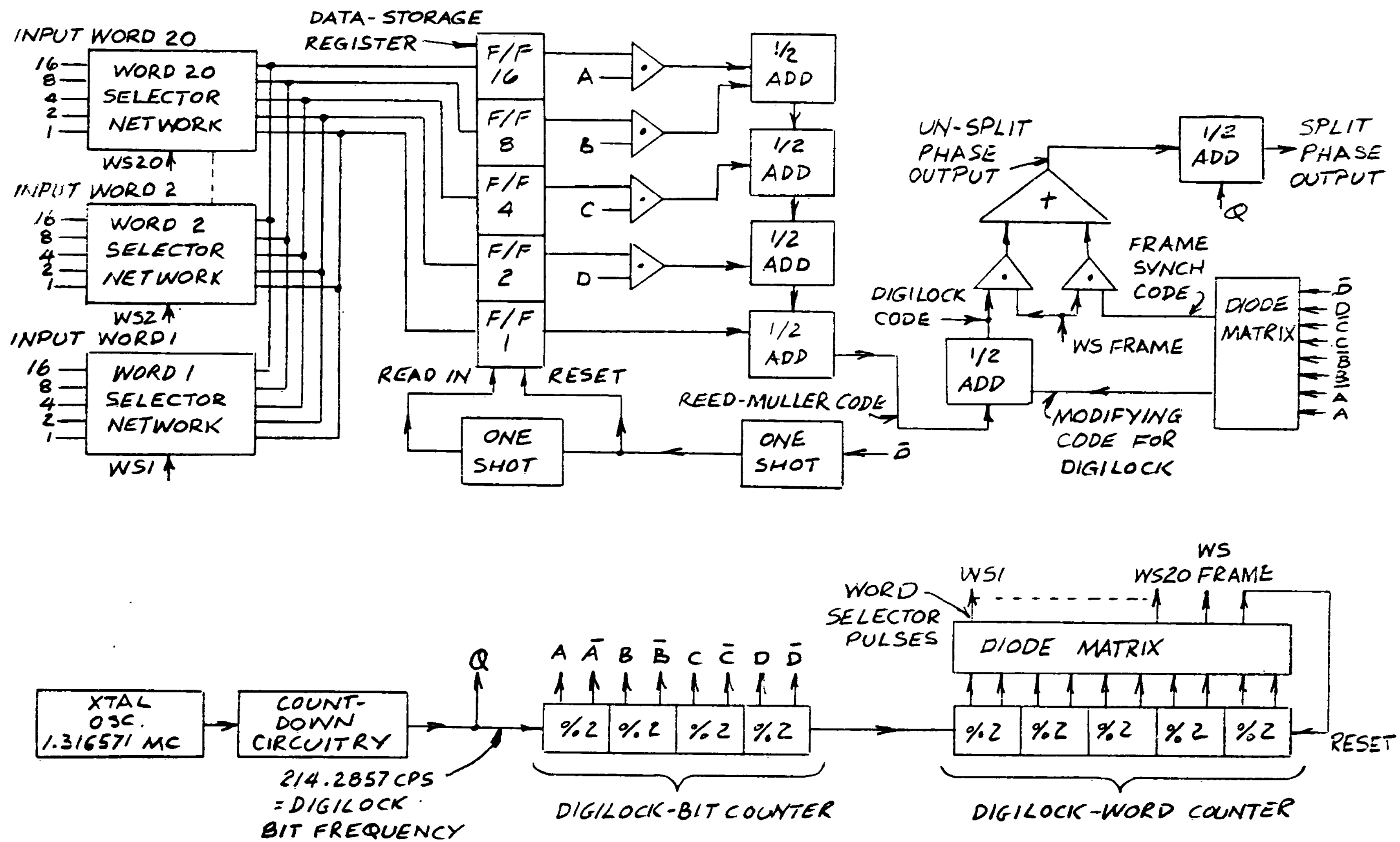
FLIGHT UNIT



DIGILOCK GROUND STATION



DIGILOCK DATA-REDUCTION STATION



BLOCK DIAGRAM OF DIGILOCK ENCODER
(64 BITS PER SECOND INFORMATION RATE)

TABLE 1 16-Bit Codes for Data Words

5-bit data word	16-bit output word			
00000	1101	1111	0111	1001
00001	0010	0000	1000	0110
00010	1101	1111	1000	0110
00011	0010	0000	0111	1001
00100	1101	0000	0111	0110
00101	0010	1111	1000	1001
00110	1101	0000	1000	1001
00111	0010	1111	0111	0110
01000	1110	1100	0100	1010
01001	0001	0011	1011	0101
01010	1110	1100	1011	0101
01011	0001	0011	0100	1010
01100	1110	0011	0100	0101
01101	0001	1100	1011	1010
01110	1110	0011	1011	1010
01111	0001	1100	0100	0101
10000	1000	1010	0010	1100
10001	0111	0101	1101	0011
10010	1000	1010	1101	0011
10011	0111	0101	0010	1100
10100	1000	0101	0010	0011
10101	0111	1010	1101	1100
10110	1000	0101	1101	1100
10111	0111	1010	0010	0011
11000	1011	1001	0001	1111
11001	0100	0110	1110	0000
11010	1011	1001	1110	0000
11011	0100	0110	0001	1111
11100	1011	0110	0001	0000
11101	0100	1001	1110	1111
11110	1011	0110	1110	1111
11111	0100	1001	0001	0000
Frame Sync	0001	0000	0010	0000

APPENDIX C

TYPICAL SLV-1B PAYLOAD WEIGHTS WITHOUT EXPERIMENT

Telemetry Components		
Digilock Encoder	1.1	
Transmitter (250 mw)	.44	
Antenna System	.47	
Misc. Relays, wiring, plugs	.5	
		2.51
Batteries (For experiment drawing .5 amp @ 28V)		
18 PM-5 (+28V, 7.5 amp-hr)	5.2	
5 PM-2 (+7.5V, 2 amp-hr)	.78	
4 PM-1 (-6V, 1 amp-hr)	.28	
Battery cans and mounts	.66	
		6.92
Mounting Hardware		
Foam (milled out, with brackets)	1.32	
Magnesium-alloy payload mounting plate	1.5	
RF - Thermal shield, mounting flange, connector cover	.76	
Bonding compound	.4	
Dynamic balance weight	.8	
		4.78
Vehicle Performance Instrumentation		
Longitudinal accelerometer and VCO	.4	
Lateral accelerometer and VCO	.4	
Time Delay Relay	.19	
		.99
Radar Augmentation System		.5